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# PHASE 1A STUDY REPORT

# VOYAGER SPACECRAFT

## VOLUME 7
## 1969 FLIGHT TEST SPACECRAFT AND OSE

30 July 1965

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# I. INTRODUCTION

This volume serves to consolidate the presentation of spacecraft and OSE design as well as program planning for a 1969 flight test mission. Because of the high degree of similarity between the system designs for the 1969 and 1971 systems a great deal of the basic descriptive data is available elsewhere in the Phase IA study report and is therefore only summarized here and included by reference.

Presentation of the spacecraft design relies heavily on material in Volume 2 which is organized into individual design documents and identified by a VS number. This is also the case for the OSE design in relation to the presentation of the 1971 version in Volume 6. Program implementation has been defined and presented in terms of an integrated plan in Volume 3 and that volume serves as the detailed source for such information for the 1969 mission.

# II. MISSION OBJECTIVES AND ALTERNATIVES

The over-all objective of the 1969 test flight mission is "to achieve improved probability of 1971 mission success." The nature and role of flight testing must be taken into account in considering this objective. That is, flight testing should not be considered as competing with or replacing ground testing. Specific investigations under controlled conditions are generally better done by ground test programs, whereas flight testing should represent confirmation testing to bring out any unknown effects not realizable under ground test conditions. Of course flight testing should provide significant data regarding any undesired spacecraft behavior if such occurs, so that remedial action can be taken.

When the proper role of flight testing is considered, it seems clear that to enhance the probability of 1971 mission success, the 1969 flight test should utilize a spacecraft that incorporates the 1971 design to a maximum extent. Furthermore, the totality of operations from manufacture through launch as well as spaceflight operations should proceed along lines as close as possible to those for the 1971 mission.

When one considers (1) an earth orbiter mission, (2) an escape mission that does not go to Mars, and (3) a Mars flyby mission, these must be evaluated in terms of how far one can go toward achieving similarity between a corresponding 1969 system and the 1971 mission. As the latter provides a higher degree of appropriate operational experience, clearly it is to be selected if the associated increased schedule and payload limitations do not cause an overbalancing reduction in the applicability of the spacecraft design. The Phase IA Study has shown that a valid test version of the 1971 spacecraft for a Mars flyby trajectory can indeed be realized within the constraints of the 1969 mission. This conclusion applies to the program and schedule constraints as well as those for the spacecraft design. Hence the Mars flyby mission becomes an obvious choice.

The basic 1969 spacecraft design is not affected directly by the choice of a Mars flyby mission. This design, as described in following sections, is primarily responsive to the criterion of achieving a maximum applicability and identity with the 1971 system. There is a significant weight margin associated with the basic design, however, that is available for various test options or for science payload, and the use of this margin does become closely linked to the specific nature of the mission. It is only the Mars flyby mission that offers the possibility to use this available weight for science objectives that at the same time can serve to increase the test applicability so as to validate the important 1971 spacecraft science interface area.

As possibilities for the 1969 flight test become better defined, it is clear that the major area of difficulty arises in regard to schedule and program implementation rather than in defining a valid spacecraft test design. Also, it is vital in achieving the payoff from such a flight test to avoid the creation of two separate teams for the two missions. All of the Phase IA program planning has taken as a tenet the achievement of a single integrated program. Such a program has been defined and is presented in Volume 3.

An obvious yet very significant point is that the use of a Saturn Centaur launch vehicle in 1969 would not only yield the ultimate return in enhanced probability of success for the 1971 mission, but would bring about major relief in the program implementation area. Such a program would avoid the double load in structures and detailed spacecraft mechanical and electrical integration and test that is inherent in the program utilizing the two separate vehicles. This is true even though there is essentially only a single subsystem development effort.

## III. MISSION CONSTRAINTS AND DESIGN CRITERIA

The degree of achievement of similarity between the 1969 flight test spacecraft and the 1971 spacecraft is limited by the need to satisfy certain mission constraints and design criteria. In a general sense the same criteria apply to the 1969 design as to the 1971 system. Additional Atlas-Centaur peculiar items are discussed below.

### 1. LAUNCH VEHICLE

For purposes of the present discussion the launch vehicle for the 1969 mission is to be the Atlas-Centaur. The mission plan calls for two space vehicles to be launched. Each is to incorporate a 1969 flight test spacecraft, with these being essentially identical.

### 2. LAUNCH VEHICLE PERFORMANCE

The most significant constraint for the 1969 mission is the use of an Atlas-Centaur instead of the Saturn-Centaur used for the 1971 mission. At a value for $C_3$ (twice the injection energy) of 15 $km^2/sec^2$ the Saturn-Centaur has a payload capability of over 8000 pounds as compared to about 1500 pounds for the Atlas-Centaur. The required $C_3$ for the 1969 test flight depends on trajectory considerations as discussed in IV. 1. The allowable spacecraft weight is determined as a function of $C_3$ by the payload capability curve of Figure 3-1.

### 3. SPACECRAFT ENVELOPE

The spacecraft is required to fit forward of the Centaur and within the nose fairing as shown in Figure 3-2. The original envelope constraint allows an additional cylindrical length to be considered, but no increase in diameter. In keeping with this, a 42-inch extension to the cylindrical section of the Centaur has been incorporated into the allowable spacecraft envelope shown in Figure 3-3.

### 4. LAUNCH COMPLEX

AFETR facilities at Cape Kennedy, Florida, will be used for Atlas/Centaur launch and prelaunch operations. The launch complex equipment configuration is to be compatible with the requirements and

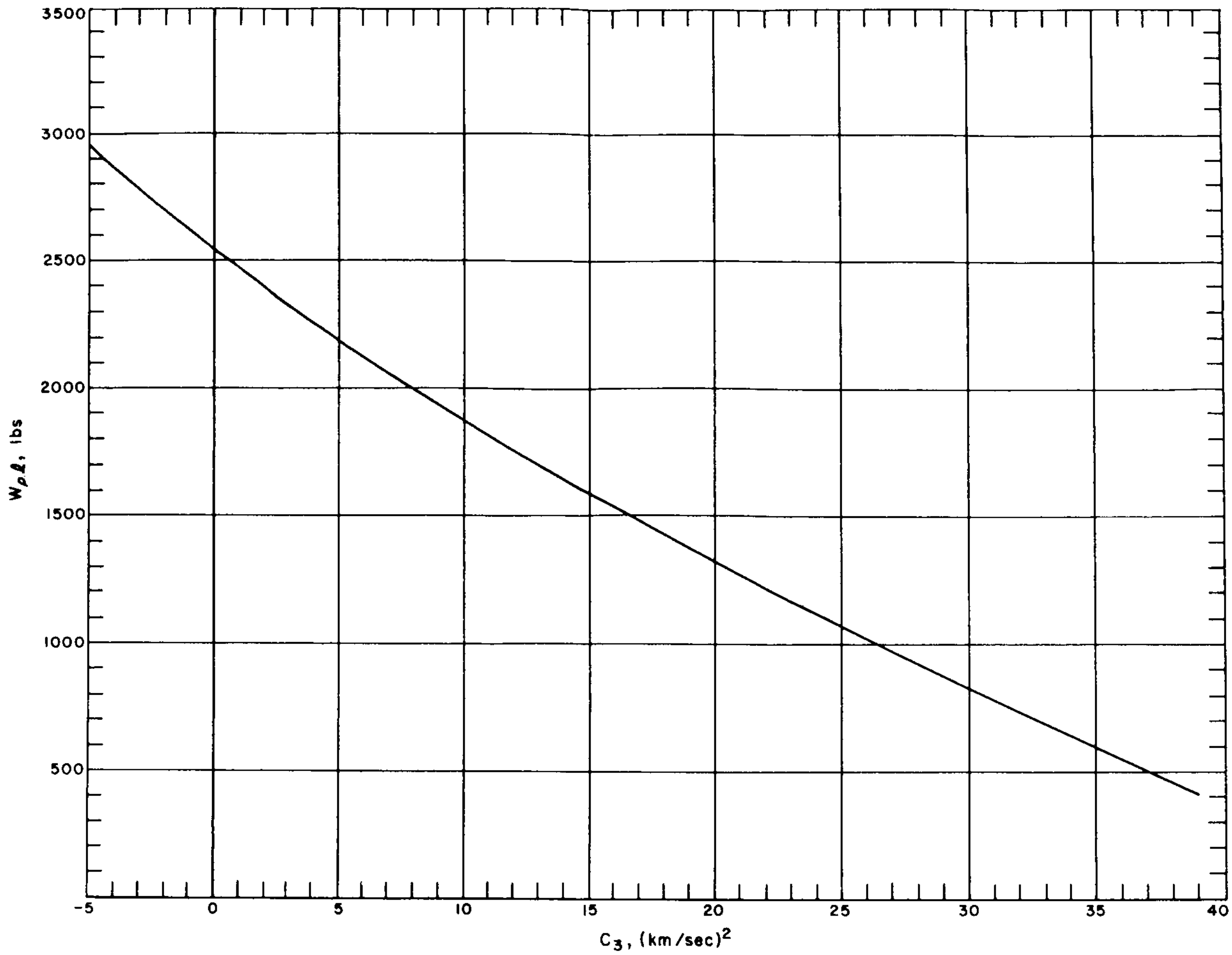



Figure 3-1. Atlas-Centaur Payload Capability for Mars Mission

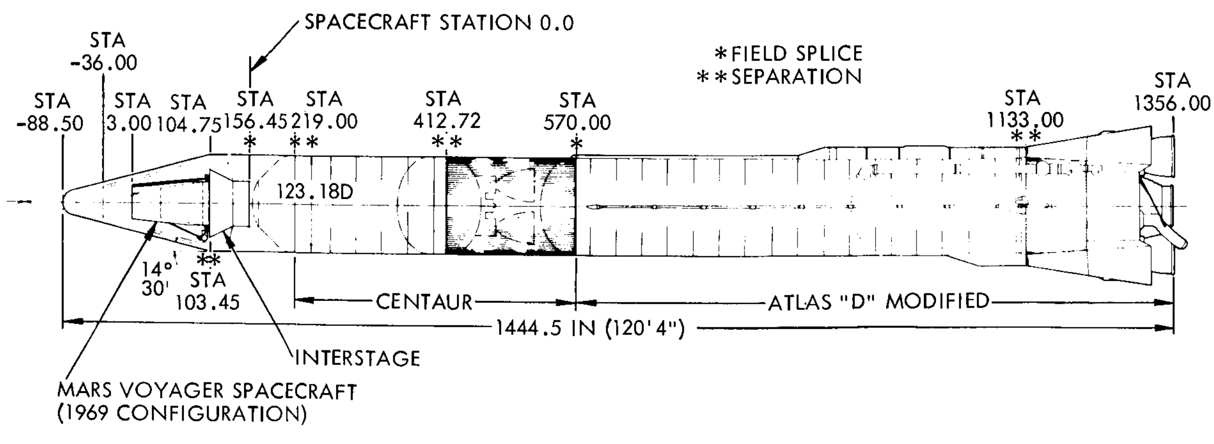



Figure 3-2. 1969 Voyager Launch Configuration

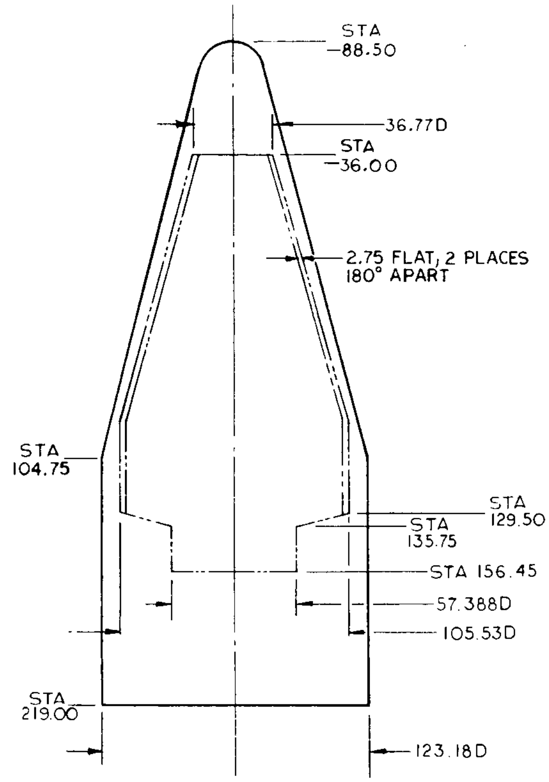



Figure 3-3. 1969 Voyager Spacecraft Allowable Envelope

restraints of AFETR Launch Complex 36 A and B, except that the OSE design philosophy and requirements are not to be compromised by use of the existing Complex-36 hardware. However, existing Complex-36 items such as multiconductor long lines, facility power, rack-mounting areas and space, air conditioning, etc., is to be considered in the LCE design and in prelaunch testing. Use of existing Complex-36 patch panels, cable junction boxes, etc., will not be mandatory and these will be used only if the added number of circuit interconnections and increased potential of cross-talk and interference can be demonstrated in advance of field operations to have no detrimental effect on LCE system operations.

## 5. LAUNCH OPERATIONS

Prelaunch assembly and checkout will be conducted in Hangar H or J for the Atlas and Centaur stages and at the spacecraft checkout facility (Building AO) for the spacecraft. An explosive safe facility will be used for propellant and gas loading, final spacecraft alignment, installation of other hazardous components, and spacecraft encapsulation.

The two upper portions of the Centaur nose fairing, which are half-cones, are to be installed on the upper spacecraft adapter, with spacecraft attached, in the hangar at AFETR. Following this installation, the encapsulated spacecraft is not physically accessible. The encapsulated spacecraft, including the upper spacecraft adapter, is then mated to the lower adapter section of the Centaur at the launch stand. Prior to mating the spacecraft to the Centaur, the two lower-half cylindrical sections of the nose fairing are then installed between the upper portions and the Centaur.

## 6. TRAJECTORY

Trajectories are to be compatible with Cape Kennedy as a launch site utilizing Atlas-Centaur as the launch vehicle. A launch period of at least 30 days is to be provided and a parking orbit ascent is to be utilized. An arbitrary limit of a 25-min parking orbit has been established for planetary missions; hence all vehicle equipment and expendables will be sized for this duration, and all performance calculations will be based upon this limitation. Any request for a parking orbit exceeding 25 minutes is to be submitted to JPL for evaluation and action, as required. Additional trajectory constraints are as follows:

<u>Trajectory Constraints</u>

| | |
|---|---|
| Launch azimuth ($\sigma_L$) | 90 to 114 deg |
| Injection true anomaly ($\eta_i$) | +4 deg |
| Parking orbit altitude | 90 nm |

For Mars fly-by mission either Type I or II are allowed.

## 7. INJECTION ACCURACY

The current projected estimate of the 1-sigma midcourse velocity to correct miss plus time of flight for the 1969 Mars mission is 10 to 15 meters/sec.

## 8. LAUNCH VEHICLE—SPACECRAFT INTERFACE

Centaur payload support items will be held to a minimum. As a specific example, wideband telemetry through the Centaur telemeter will not be required. Although the possibility for special signals from the Agena-type timer exists, the electrical interface are to be kept to an absolute minimum and special signals will not be allowed.

The physical interface between the launch vehicle and the flight spacecraft is the field joint between the spacecraft adapter and the Centaur mounting structure. This interface will include a mechanical joint and electrical connectors, including connectors for any functions between the spacecraft and Centaur and for the spacecraft (umbilical) connection to the launch complex equipment. No other physical connection is required.

Conditioned are circulated internally to the nose fairing is available for temperature control of the spacecraft during on-stand operations. The air temperature, humidity, flow rate and direction of flow are optional.

At launch, when the air conditioning is disconnected, the air which is inside the nose fairing cavity escapes through small holes near the lower end of the nose fairing. The resulting ambient pressure is as defined in Figures A-3 and A-4 of the Mission Guidelines.

With regard to aerodynamic heating, the nose fairing on a worst case (that is, a 3-sigma maximum heating trajectory) has a heating rate of 157 Btu/hr/sq ft, which is considered to be the average for the period of time from launch through nose fairing ejection.

The center-of-gravity limitation of the payload atop the Centaur is a cylinder 1 inch in radius, with ceterline on the vehicle roll axis, and with the ends of the cylinder at launch vehicle stations 95 and 150. Refer to Figure 3-2 for vehicle station locations.

# IV. SYSTEM DESCRIPTION

## 1. TRAJECTORY CONSIDERATIONS

Trajectory considerations for Mars missions have a strong effect on system design when scientific objectives and the operation of science experiments are taken into account. From the viewpoint of testing the basic 1971 spacecraft design, such considerations are mainly eliminated so that the effect on allowable spacecraft weight becomes the primary factor of interest. This has been discussed below along with several trajectory characteristics related to the 1971 spacecraft design.

### 1.1 Trajectory Type

The constraints affecting the selection of a trajectory for the 1969 mission have been given in Section III. For a Mars fly-by mission, the data given there relating to launch azimuth, true anomaly at injection, and the 25 minute coast time limit can be combined for purposes of the present discussion into a restriction on the declination of the trajectory launch asymptote (DLA). This condition can be expressed as

$$-33^{\circ} < \mathrm{DLA} < +10^{\circ}$$

In keeping with the basic criterion to achieve a high degree of similarity between the 1969 design and that for the 1971 spacecraft, it has been found from configuration studies that a separated spacecraft weight of 1400 pounds is very desirable. From Figure 3-1, this amounts to a value for the energy quantity $C_3$ of approximately 18 $km^2/sec^2$. This energy level in conjunction with a minimum launch period requirement of 30 days eliminates consideration of Type I earth-Mars trajectories. Specifically, the longest launch period available in which both $-33^{\circ} <$ DLA and $C_3 < 18$ $km^2/sec^2$ are satisfied is about 20 days. As we are interested in accommodating more rather than less weight relative to the 1400 pounds, Type I trajectories are considered no further at this time. This is not a significant conclusion, however, as the present purpose is merely to establish that suitable allowable weight is available for a meaningful 1969 mission, and this will be done below in terms of Type II trajectories.

## 1.2 Allowable Weight

The most important effect that trajectory selection has on spacecraft design is the determination of an allowable spacecraft separated weight. We now consider this important item. Turning attention to Type II trajectories, Figure 4-1 illustrates some of the important parameters for such trajectories plotted as contours on the plane having coordinates as earth launch date and Mars arrival date. The DLA restriction is not binding for the region of interest. It is seen from Figure 4-1 that long launch periods may be provided which satisfy the constraints, and permit payloads to be considered to over 1800 pounds ($C_3 \simeq 10$). When we impose the consideration of earliest launch date satisfying the requirement for a payload of over 1400 pounds (or $C_3 \leq 18$ km$^2$/sec$^2$) we obtain the following results of interest:

| Weight | Earliest Launch Date | Earliest Arrival Date |
|---|---|---|
| 1400 | Dec. 22, 1968 | Sept. 7, 1969 |
| 1500 | Dec. 28, 1968 | Sept. 15, 1969 |
| 1600 | Jan. 5, 1969 | Sept. 29, 1969 |
| 1700 | Jan. 15, 1969 | Oct. 14, 1969 |

Launch periods of over 100 days may be chosen for these weights, so that the weight constraint does not proscribe the end of the launch period.

Hence, it appears that 1400 pounds represents a conservative estimate of spacecraft separated weight for 1969 missions. This is discussed further in IV. 5.

## 1.3 Communication Distance

One of the important trajectory related design parameters from the 1971 system is a maximum communication distance to earth of $390 \times 10^6$ km. In keeping with the 1969 flight test objective of validating the 1971 design, it therefore appears desirable to evaluate the spacecraft communication subsystem out to this design limit. The achievement of such a condition depends on both the earth-Mars trajectory and the distance and direction

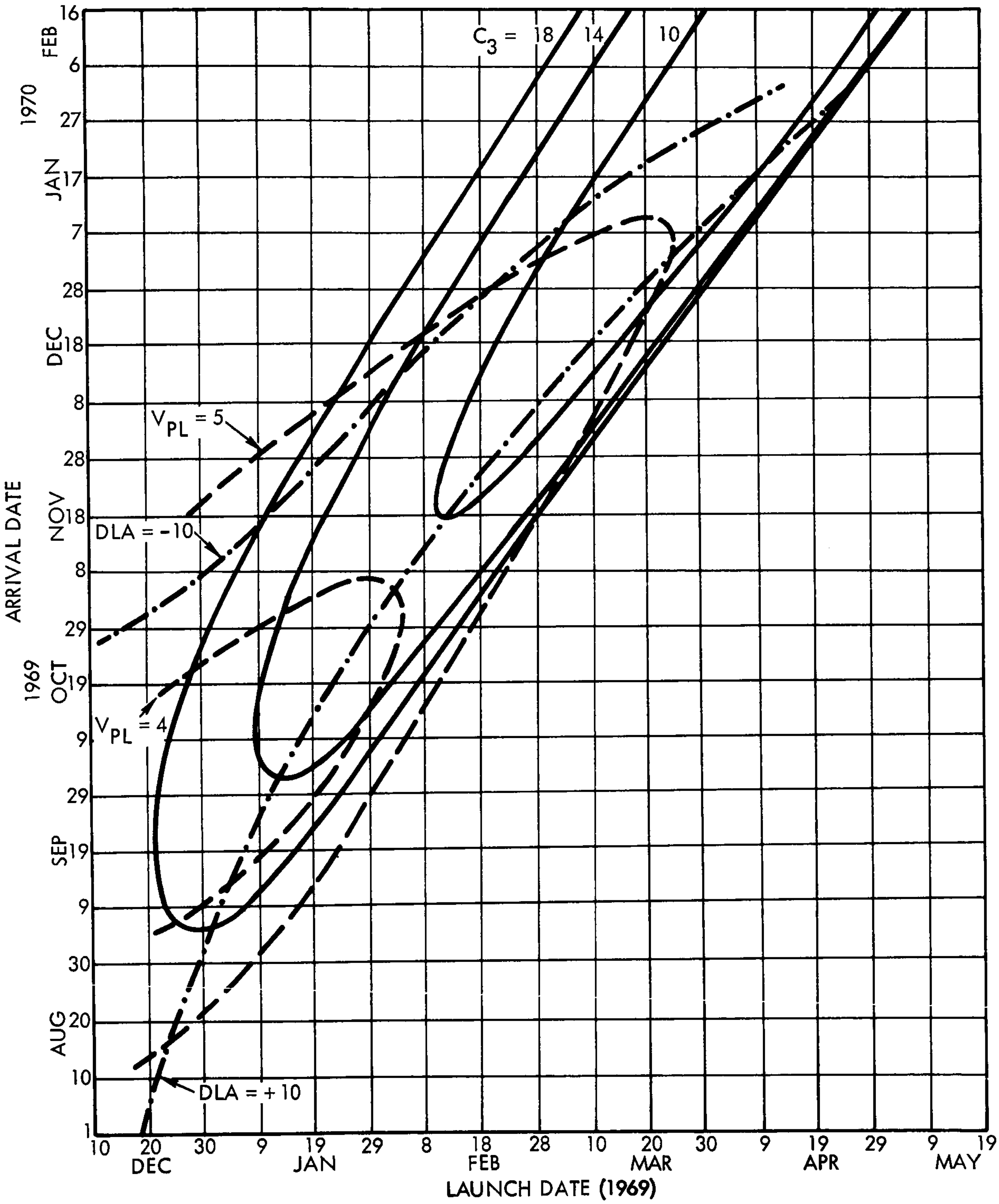



Figure 4-1. Characteristics of 1969, Type II, Earth-Mars Trajectory

from Mars at the spacecraft's closest approach, as the subsequent spacecraft trajectory is affected by these. Without a detailed study, accurate predictions can not be made. However, a rough estimate indicates that the maximum distance from the earth attained within 3 years of launch will be in the range 325 to 375 $\cdot 10^6$ km, and it will occur approximately 6 to 8 months after encounter, or 15 to 18 months after launch. Thus a test at the maximum communication distance cannot be achieved on a very timely basis. This is not felt to represent any serious limitation to the validity of the 1969 test, however, as performance of the communications system can be evaluated at smaller distance and extrapolated to the design limit.

## 1.4 Solar Distance

Another trajectory-related design parameter for the 1971 system is the solar array performance at a distance of 1.67 AU from the sun. Achievement of such a distance from the sun would have to be accomplished during the earth-Mars transfer, or after encounter. This is because Mars is only 1.38 to 1.45 au from the sun at arrival dates for the 1969 window. For the spacecraft to achieve 1.67 au from the sun before arriving at Mars, it would have to be on a very late-arriving trajectory which loops out to 1.67 au before returning to about 1.45 au for encounter. This is within the capability of the launch vehicle, as a $C_3$ of 12 $km^2/sec^2$ is enough for an earth-launched spacecraft to get to 1.67 au from the sun by a Hohmann transfer. However, the mission times are large, with some 8 to 9 months required to get out to aphelion, and another 4 months to return to Mars. For a typical trajectory of this type the spacecraft is launched in May 1969, reaches aphelion at 1.67 au from the sun in January 1970, and encounters Mars in May 1970.

To achieve 1.67 au from the sun after encounter, it would be desirable to use the earliest encounter dates. For trajectories arriving at Mars later, the spacecraft is already returning toward the sun, and deflection of its trajectory by Mars is not likely to redirect it outward enough to reach 1.67 au. Even for early arrivals, the radial component of spacecraft heliocentric velocity at encounter is only slightly positive. It would then require maximum deflection by Mars gravity to increase this component sufficiently to achieve 1.67 au at ophelion. If this method succeeds it requires early launch (Dec., 1968), early encounter (Sept. 1969) with passage to the east of Mars, and arrival at aphelion about January, 1970.

Thus it is possible to achieve 1.67 au from the sun around January, 1970, by two routes: 4 months before a very late arrival, or 4 months after a very early arrival. Neither method appears very attractive. A suitable alternative is to simulate increased distance from the sun by tipping the spacecraft so that the sun's rays do not have normal incidence with the solar array. At 1.4 au, an angle of $44^{\circ}$ is required to simulate 1.65 au. This angle could be maintained as in maneuvers. Since one panel does not have louvers, solar incidence on this panel will not affect thermal control significantly.

## 1.5 Mars Eclipse

The eclipse condition during orbital operations at Mars represents an important design consideration because of low temperature in the solar array for long duration eclipses. The question then arises as to whether it is possible in the 1969 mission to achieve an eclipse suitable time to evaluate the temperature predictions for 1971. Eclipses at Mars for flyby missions are generally of short duration. Since the velocities at which the spacecraft passes Mars are in the range 4 to 6 km/sec, and the shadow has a maximum width of 6800 km, the eclipse times range from 19 to 28 minutes for a spacecraft crossing the shadow almost perpendicularly. The smaller velocities (and longer eclipses) are achieved only by not passing too close to the planet. Also, the eclipse time may be lengthened somewhat by having the spacecraft cross Mar's shadow diagonally. Unfortunately, this requires a late arrival date, and for these arrivals the approach velocity increases so as to cancel some of the effect from the diagonal approach. Table 4-1 indicates approximate eclipse times for three different earth-Mars trajectories. In each case, it is assumed that passage is to the west, and through the fullest part of the shadow.

Table 4-1

Maximum Eclipse Times (Minutes)

| | | | |
|---|---|---|---|
| Launch date | Jan 9, 1969 | Mar 25, 1969 | Mar 7, 1969 |
| Arrival date | Oct 4, 1969 | Jan 5, 1970 | Feb 12, 1970 |
| ZAP angle | $90^{\circ}$ | $45^{\circ}$ | $30^{\circ}$ |
| $V_{\infty}$, km/sec | 3.8 | 5.0 | 6.0 |
| Close passage (2000 km alt) | 20 | 24 | 32 |
| Far passage (over 20,000 km alt) | 28 | 33 | 38 |

A 38 minute eclipse is about the longest that can be achieved by a flyby mission for a 1969-Type II opportunity, and it is characterized by long transit time, late arrival, and a $C_3$ equal to 18 $km^2/sec^2$, restricting payload to 1400 pounds. Therefore, matching orbital eclipse design times is not feasible in 1969. However, it appears possible to simulate eclipse conditions adequately by turning the solar array edgewise to the sun and to obtain a satisfactory evaluation of solar array thermal effects in this way.

## 2. CONFIGURATION

The 1969 flight test spacecraft has been designed in accordance with the criteria and constraints presented in Section III. In short, this amounts to achieving the maximum similarity to the 1971 spacecraft design within the Atlas-Centaur weight and space constraints. The same general design concepts and guidelines presented in VS-3-110, Volume 2, for the 1971 design are applicable to the 1969 configuration.

## 2.1 General Arrangement

The configuration and layout of equipment for the 1969 spacecraft is shown as an inboard profile in Figure 4-2 and as an outboard profile in Figure 4-3. The installation geometry for equipment such as sensors, antennas, and reaction jets is shown in Figure 4-4.

To indicate the potential of the 1969 spacecraft, several possible science and test installations have been indicated in Figure 4-2:

a) The main views of the drawing show (in phantom) the Mariner photographic science package mounted on the forward end of the spacecraft.

b) A partial view at the bottom of the drawing indicates a test installation for the double gimbaled planet-oriented package of the 1971 spacecraft.

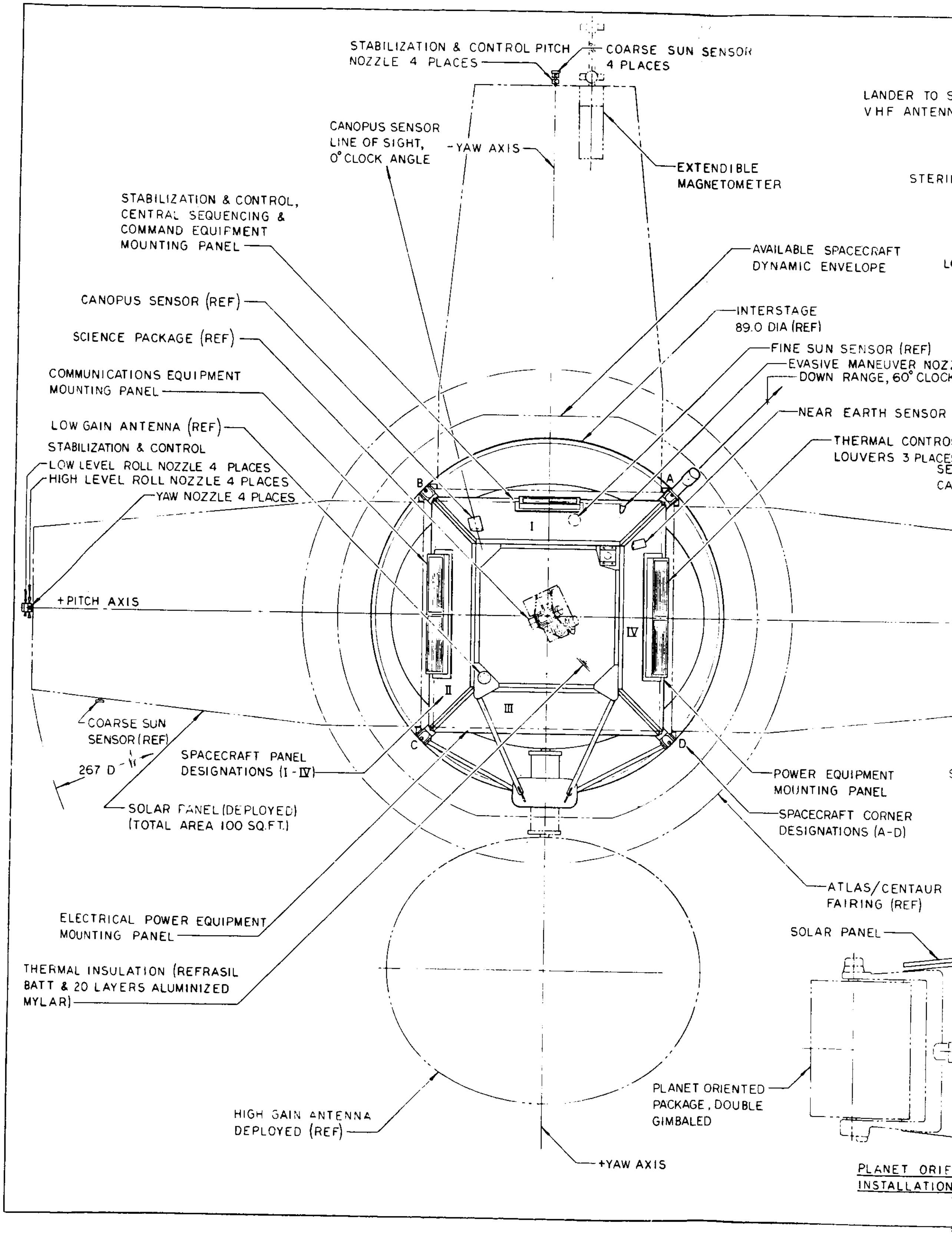
c) An auxiliary view at the top of the drawing indicates a planetary probe installation that could be carried on the 1969 test flight. Here a 2-foot diameter atmospheric probe of up to 100 pounds could be launched into the Martian atmosphere. The VHF antenna for the lander to spacecraft radio link would mount to a solar panel as indicated on the drawing.

d) An extendible magnetometer experiment is shown installed on one of the solar panels.

A test installation involving the exact separation nut and bolt catcher elements, initiators and firing circuitry as used on the 1971 spacecraft capsule interface has been installed on the forward end of the spacecraft in order to test the 1971 design.

In general the arrangement of the spacecraft is symmetrical, with the deployable items and consumables located on or close to the centerline. This arrangement minimizes structural weight, provides for ease of center of mass control and minimizes solar torque unbalance.

Four corner posts, or longerons, similar to the ones used on the 1971 configuration carry the boost loads in the panels comprising the equipment compartment portion of the spacecraft, which has the geometry of a four-sided truncated pyramid. Four-sided frames are located at the forward and aft ends of the spacecraft. The forward and aft micrometeoroid


STABILIZATION & CONTROL PITCH NOZZLE 4 PLACES
COARSE SUN SENSOR 4 PLACES
LANDER TO S
VHF ANTENN
CANOPUS SENSOR LINE OF SIGHT, 0° CLOCK ANGLE
-YAW AXIS
EXTENDIBLE MAGNETOMETER
STABILIZATION & CONTROL, CENTRAL SEQUENCING & COMMAND EQUIPMENT MOUNTING PANEL
AVAILABLE SPACECRAFT DYNAMIC ENVELOPE
CANOPUS SENSOR (REF)
INTERSTAGE 89.0 DIA (REF)
SCIENCE PACKAGE (REF)
FINE SUN SENSOR (REF)
COMMUNICATIONS EQUIPMENT MOUNTING PANEL
NEAR EARTH SENSOR
LOW GAIN ANTENNA (REF)
STABILIZATION & CONTROL
LOW LEVEL ROLL NOZZLE 4 PLACES
HIGH LEVEL ROLL NOZZLE 4 PLACES
YAW NOZZLE 4 PLACES
+PITCH AXIS
COARSE SUN SENSOR (REF)
267 D
SPACECRAFT PANEL DESIGNATIONS (I - IV)
SOLAR PANEL (DEPLOYED) (TOTAL AREA 100 SQ.FT.)
POWER EQUIPMENT MOUNTING PANEL
SPACECRAFT CORNER DESIGNATIONS (A-D)
ATLAS/CENTAUR FAIRING (REF)
ELECTRICAL POWER EQUIPMENT MOUNTING PANEL
SOLAR PANEL
THERMAL INSULATION (REFRASIL BATT & 20 LAYERS ALUMINIZED MYLAR)
PLANET ORIENTED PACKAGE, DOUBLE GIMBALED
HIGH GAIN ANTENNA DEPLOYED (REF)
+YAW AXIS

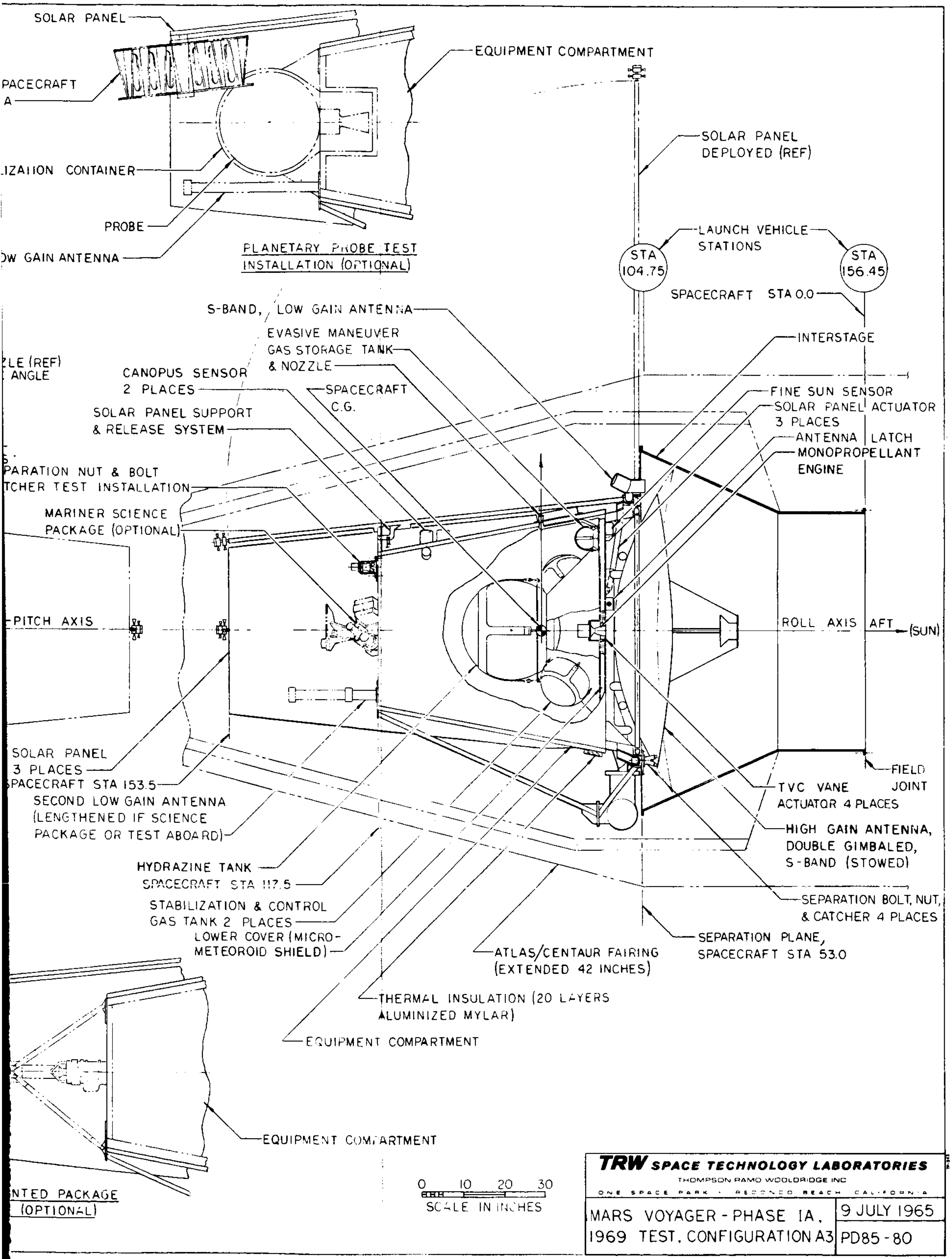



Figure 4-2. 1969 Voyager Spacecraft
Inboard Profile



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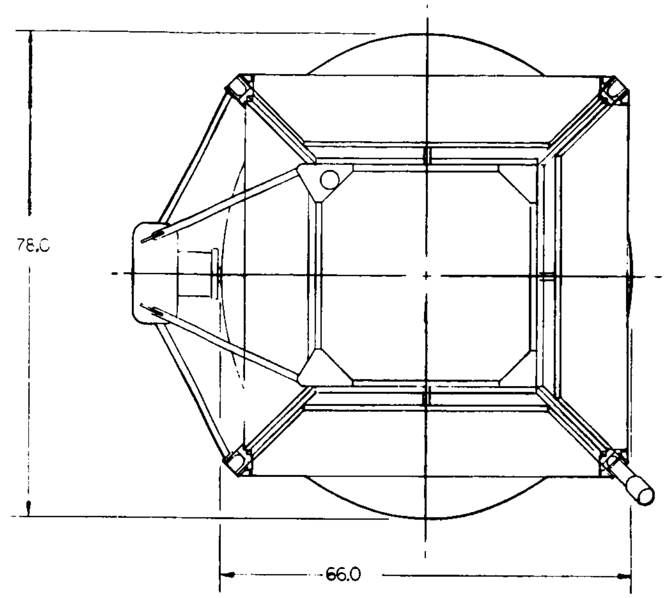


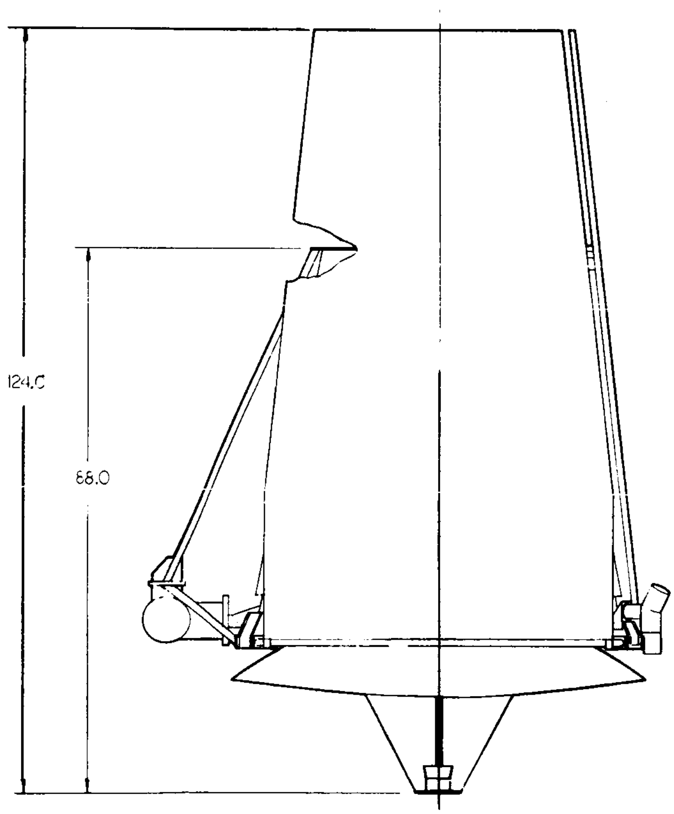
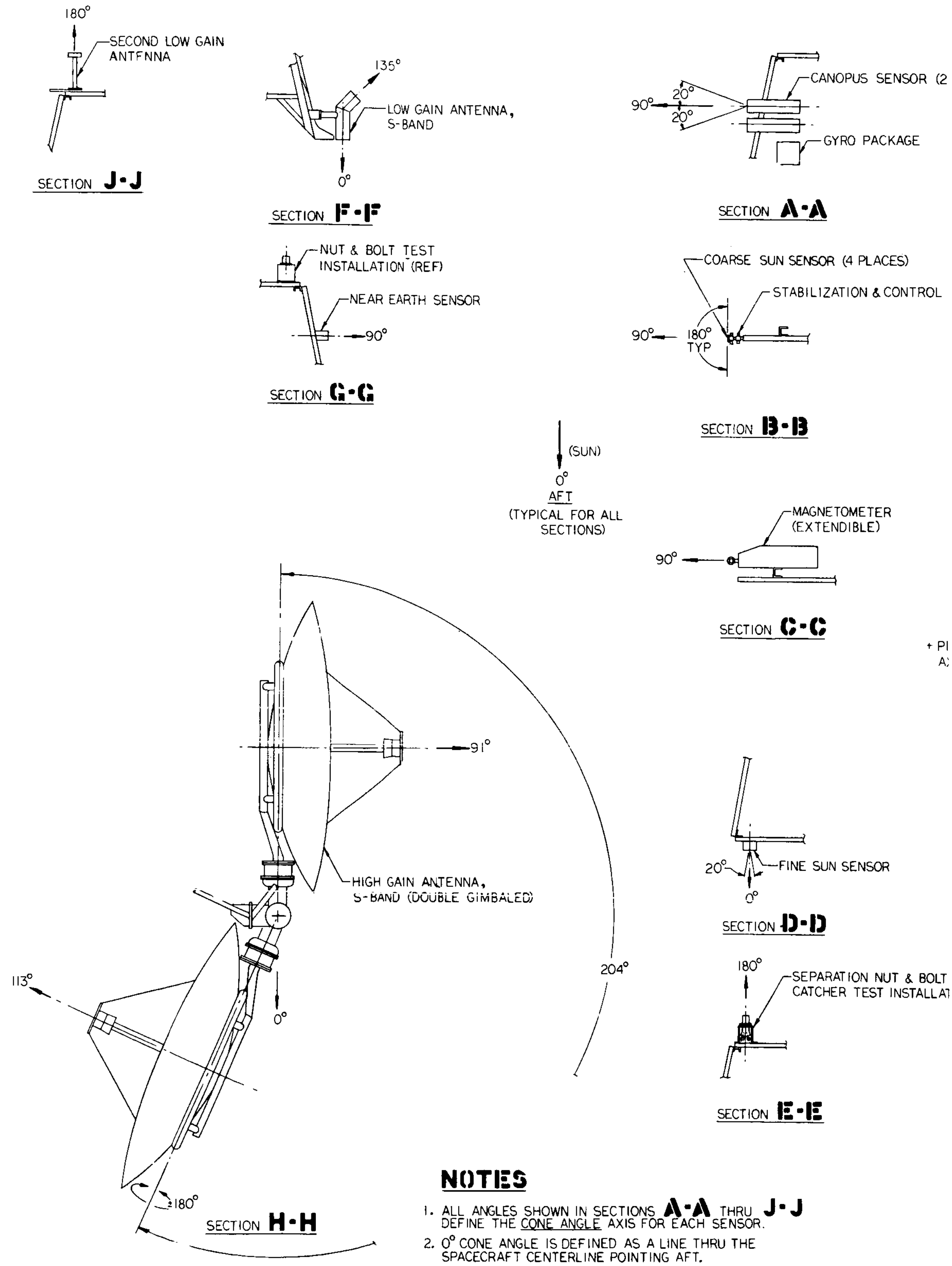




Figure 4-3. 1969 Voyager Spacecraft Outboard Profile




## NOTES

1. ALL ANGLES SHOWN IN SECTIONS **A-A** THRU **J-J** DEFINE THE CONE ANGLE AXIS FOR EACH SENSOR.
2. 0° CONE ANGLE IS DEFINED AS A LINE THRU THE SPACECRAFT CENTERLINE POINTING AFT.
3. SECTIONS SHOWN HAVE BEEN ROTATED FOR CLARITY.

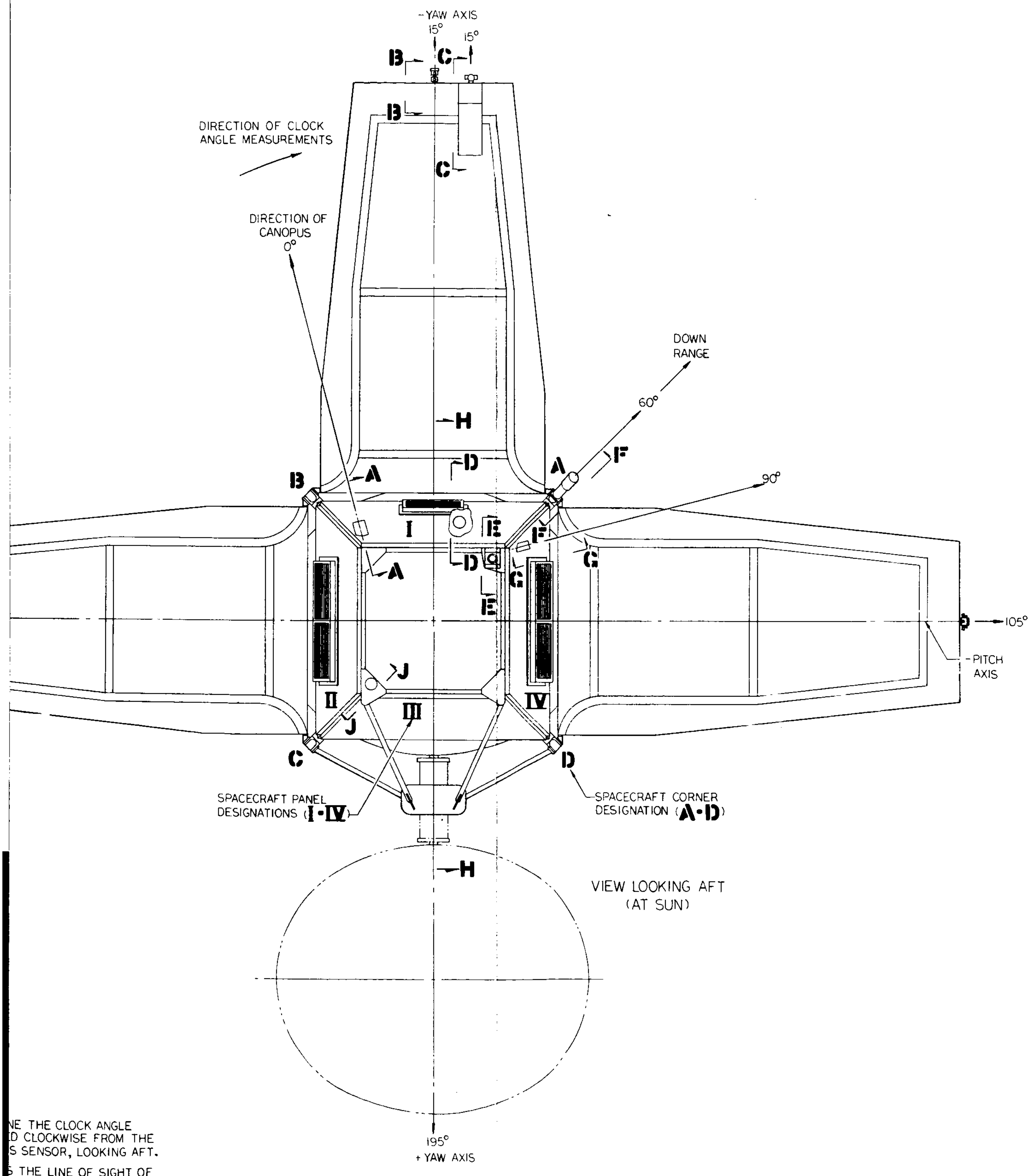



Figure 4-4. 1969 Voyager Spacecraft Sensor Geometry

2

and radiation protection covers attached to these frames complete the basic spacecraft structure. All surfaces of the spacecraft except the radiating areas covered by the thermal control louvers are thermally insulated. The solar panels shade the thermal control louvered area from the sun.

Although it is not feasible to install and test the 1971 solid retropropulsion motor on the 1969 spacecraft, it is possible to include a midcourse propulsion subsystem in a version essentially equivalent to the 1971 design. As shown on Figure 4-2, the 1971 liquid monopropellant engine is installed on centerline at the aft end of the spacecraft. One of the 1971 internally pressurized propellant tanks, off loaded to 45 pounds of hydrazine propellant, is also located in the centerline and mounted similarly to the 1971 configuration. The engine is mounted to the aft cover which also supports the conical structure to which the midcourse propellant tank is mounted. The conical structure also supports the two stabilization and control gaseous nitrogen tanks and valving. The assemblage of the aft cover, the tankage and valving provides a modular system that is essentially the same as that for the 1971 configuration.

The three solar panels which provide the spacecraft with electric power utilize the solar panel modules and circuitry of the 1971 configuration and are sized for 100 square feet of solar array. The array can be increased to approximately 120 square feet within the confines of the fairing. The panels are stowed, during launch, against support struts and locked in place. On signal, cable cutters sever the restraining cable permitting the panel latch to release. Velocity-damped actuators force the panels to the full open position where they lock in place. Retaining lanyards prevent the severed ends of the cable from interfering with any other systems of the spacecraft.

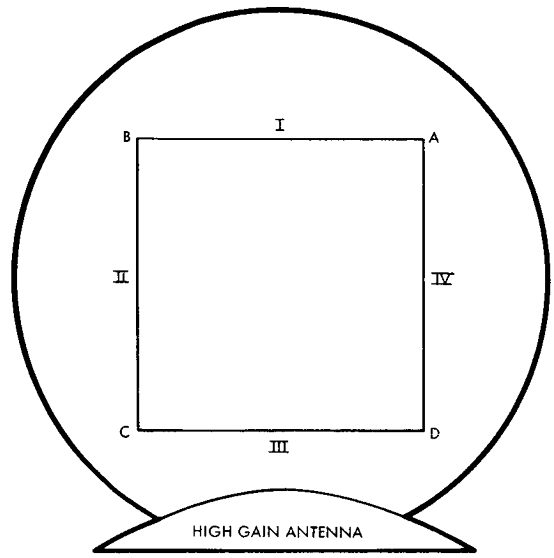
With the exception of reduced tankage and plumbing, the 1969 configuration carries the 1971 elements of the stabilization and control system. As shown in Figure 4-2, the yaw and roll nozzles are symmetrically located on the solar panels. As a symmetrical arrangement of the pitch nozzles would result in short moment arms, due to mounting directly to the equipment compartment, all pitch nozzles have been mounted to the outer end of the solar panel located at the negative yaw axis. The asymmetry is not detrimental to spacecraft operation and allows the desirable features of equal moment arms and torques, equal line losses, and thermal coupling of the lines to the panels (for increased efficiency) as obtained on the 1971 spacecraft. The stabilization and control system sensors are mounted similarly to the 1971 installation.

The 5-1/2 by 6-1/2 foot high-gain antenna (together with its double gimbal) is the same as for the 1971 configuration. The antenna is supported by struts fixed to the equipment compartment, and during launch is stowed within the interstage. It is locked in place in the same manner as for the 1971 configuration. A low-gain antenna identical to its 1971 counterpart is located at the down range "A" corner of the spacecraft. This location enhances communication coverage during powered flight. A second low-gain antenna is located on the forward end of the bus to obtain spherical coverage.

## 2.2 Electronic Equipment Packaging

A key feature in achieving similarity between the 1969 and 1971 systems is the use of the identical panels for the spacecraft equipment compartment. It has been possible to utilize four of the 1971 panels to form a compartment of square cross section with sufficient mounting area and volume for all of the spacecraft equipment, plus one panel in reserve for future requirements. Also, the resulting spacecraft mates well with the extended fairing and provides excellent accommodation for the solar array and the large high-gain communication antenna.

The three 1971 spacecraft panels with their complement of installed electronics equipment, shown as panels III, V, and VI in VS-4-550, are used essentially unchanged for the 1969 configuration. These are the CS and C/SCS, telecommunications, and power panels shown (panels I, II, and IV, respectively) in Figure 4-5. Any science electronics equipment for the 1969 mission could be accommodated easily on the blank panel III or on I, where considerable space is available. Because of the square geometry of the 1969 configuration as against the hexagonal form for 1971, it may be necessary to trim the left-hand section of the stabilization and




| PANEL NO. | TITLE | EQUIPMENT WEIGHT |
|---|---|---|
| I | STABILIZATION AND CONTROL | 30.3 LB |
| II | COMMUNICATION | 102.2 |
| III | BLANK | |
| IV | POWER | 124.1 |
| | GRAND TOTAL | 256.6 |

*REFERENCED TO AND INDENTICAL WITH
1971 MISSION PANEL LAYOUTS

Figure 4-5. 1969 Voyager Spacecraft Panel Arrangement

control equipment support rail so as to avoid interference with the telecommunications panel adjacent rail. Otherwise all panels are identical and interchangeable. The insulation and louvers are the same as for the corresponding 1971 panels.

The arrangement of a single compartment provides for maximum internal thermal coupling. It also enhances electrical distribution and minimizes the required cabling and the number of electrical connectors.

2.3 Launch Vehicle Integration

In order to accommodate a 1969 spacecraft design with the same equipment panels, the same high-gain antenna and the same general structural concept as for the 1971 spacecraft, it has been necessary to lengthen the Atlas-Centaur fairing as allowed by Section III. This extension has been incorporated into the allowable envelope of Figure 3-3. The specific extension of 42 inches embodied in the current design is somewhat arbitrary and will probably be shortened slightly when the design is refined. It has been chosen to allow a comparison with the spacecraft configuration illustrated in EPD-261.*

The mechanical interface between the spacecraft and the basic launch vehicle is the field joint located at Atlas-Centaur station 156.45 which has also been designated as spacecraft station 0.0. The number and type of spacecraft attachments to the booster are flexible and may be determined at a future date. A 53-inch long interstage structure runs from this field joint to the in-flight separation joint at Atlas-Centaur station 103.45. This interstage remains with the booster at separation. It is a semimonocoque structure composed of a cylinder for the aft part and a truncated cone for the forward part as shown in Figure 4-3. Although not shown on the drawings, large cutouts may be included in the interstage to save weight. The interstage redistributes the uniformly distributed loads at the launch vehicle field joint to four concentrated loads at four equally spaced points at the in-flight separation plane.

In-flight separation of the bus from the interstage is accomplished with the aid of separation nuts at each of the four hard points at the separation plane. These separation devices are the same as utilized for launch vehicle-spacecraft separation in the 1971 configuration. On signal, the separation nuts located on the aft side of the separation plane disengage 3/8-inch-diameter bolts and also drive the bolts out of their holes into bolt catchers mounted on the bus. The Centaur's retrothrust motors provide positive separation of the booster and spacecraft.

The electrical interface between the launch vehicle and the spacecraft includes an umbilical that is disconnected prior to launch in addition to the firing circuits for the separation devices. Except for the use of four separation devices instead of three, the 1969 design is identical to that for 1971. The mechanical, as well as the electrical, interface is shown schematically in Figure 4-6.

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*JPL Document No. EPD-261, "Mariner Mars 1969 Lander Technical Feasibility Study," 28 December 1964

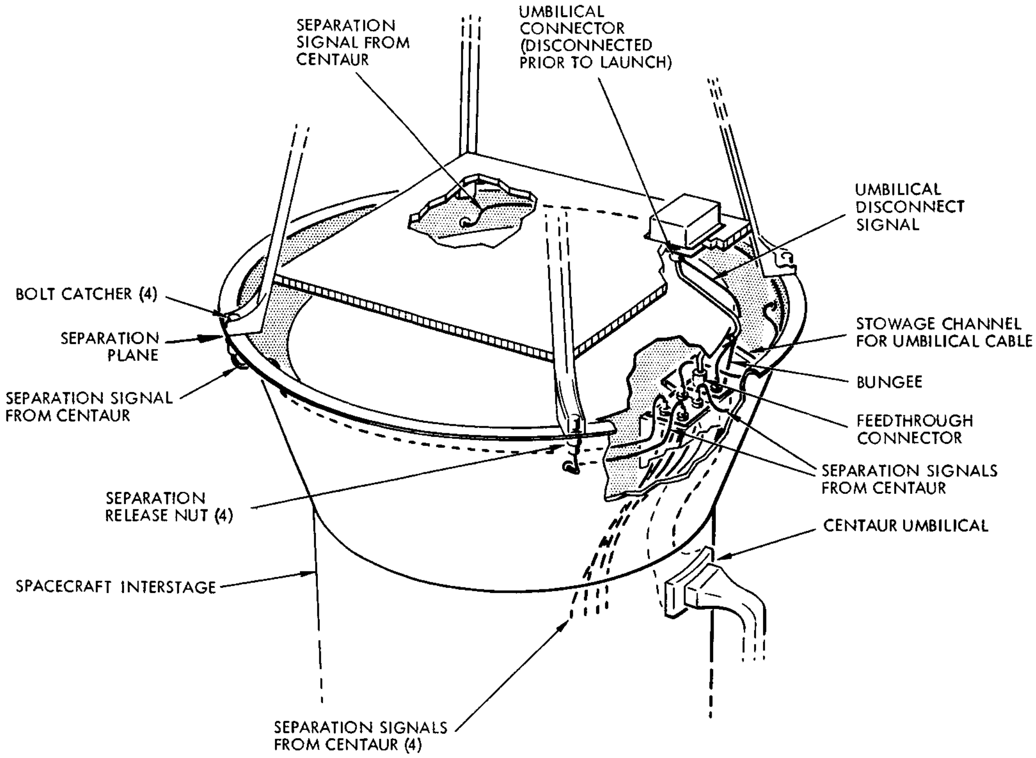



Figure 4-6. Schematic Representation of the Atlas-Centaur and 1969 Spacecraft Separation System

## 3. SYSTEM OPERATION

As brought out in the discussion in IV. 2 of the 1969 spacecraft configuration and also in the subsystem descriptions of Section V, the 1969 system is very similar to that for the 1971 mission. A corollary to this in keeping with the basic 1969 mission objective is that the 1969 system operation should be as close as possible to the operation of the 1971 system. The similarity between the two systems can be seen to provide a double payoff in that not only does it allow the hardware to be tested, but it also allows validation of prelaunch operations, the mission operations system, and the DSN. In particular, the launching of two spacecraft for the 1969 opportunity will allow the problems associated with dual spacecraft deep space missions to be worked out for the first time.

The flight sequence for the 1969 mission should be made to include all events of the 1971 mission to the maximum extent. Operations for such things as sun-Canopus acquisition and midcourse maneuvers can be carried out in the same manner as for the 1971 mission. Antenna pointing is the same as for the 1971 mission except for verification of the antenna angles before a maneuver is started. In addition it is desirable to simulate separation of the capsule vehicle maneuver, an evasive maneuver, jettisoning of the capsule adapter, and the retropropulsion maneuver. Special test installations of pyrotechnics, separation devices, etc., can serve to validate these simulated operations. An important part of such simulations is the effect of a science payload on system operations. Of course the most effective test would be to include actual science payload equipment. Of particular importance in this regard is the data automation equipment, which plays the central role in the spacecraft science payload interface. This addition is very desirable and would give an extra degree of coverage in the validation of system operation. Tests of the Mars horizon scanner could also be performed if the POP were included.

## 4. RELIABILITY

The most significant aspect of the 1969 flight test from the reliability point of view is the opportunity to subject the equipment to valid space environmental conditions for the appropriate long-term period, and to demonstrate the mission capability during actual operations. This is in keeping with the basic flight test function of design confirmation under conditions to bring out effects not realizable during ground testing. In conjunction with this over-all evaluation, it is also important to verify in detail the environments pertinent to the individual equipment and the effectiveness of redundancy management

### 4.1 Environment Verification

All projected mission reliability estimates are contingent upon accomplishment of the individual equipment reliability goals which in turn depend upon the validity of the presumed environments. This determination of the environmental conditions must be accomplished in sufficient depth and also with timeliness to allow corrective design action when found necessary. Such a determination involves establishing a valid "equivalent sustained environment" and ascertaining the absence of excessive environmental transients or gradients which could simulate unusual failure mechanisms.

The 1969 equipment environmental conditions are expected to be a valid representation of those for the 1971 mission. The early phase of powered flight involves different boosters with various environmental factors having different degrees of applicability. The Centaur induced environment is, of course, the same for both missions.

The earth-Mars transit phase is generally comparable for the 1969 and 1971 missions because of the high degree of similarity between the two configurations and the flight sequences. Specific attention will be given to effects on the spacecraft subsystems from the following environments:

a) Temperature, maximums and minimums

b) Temperature, sustained average

c) Thermal shock (rate of charge)

d) Thermal cyclic effects

e) Vacuum and decompression

f) Zero-g condition

g) Acoustic noise

h) Vibration levels and transmissibility

i) Shock

The effects of these environments, in terms of detectable failure modes and anamalous effects, will be examined in the 1969 flight as a means of assuring a more reliable flight in 1971.

## 4.2 Test of Redundancy Management

A basic nonredundant system composed of subsystem elements from the 1971 spacecraft design would be expected to have approximately a 17 per cent probability of mission success. This compares to a 71 per cent probability estimated for the design in its redundant form. Such reliability upgrading through redundant equipments is directly applicable to the 1969 spacecraft and depends upon the validity of failure sensing and equipment switching under actual use environments. For the 1969 test flight, equipment operation can be planned to simulate a wide variety of failure modes for which system response with corrective functions may be verified. In designing for failure sensing and switching (redundancy management) the use of uplink commands has been intentionally minimized. The ability of the spacecraft central sequencing command subsystem to achieve the full potential of redundancy switching in the presence of other command functions needs to be verified. Also, in those instances where digital code differences dictate which of two equipments are effective in a redundant set, the function of these selection techniques needs to be verified under actual poor signal conditions. These conditions can be simulated and validated by the test flight in 1969 so as to enhance the probability of success for the 1971 mission.

### 4.3 1969 Spacecraft Reliability

The elimination of some functions from the 1971 spacecraft configuration to form the 1969 configuration does not significantly affect the probability of mission success up to the encounter of Mars. Using the same representative mission phases as for the 1971 mission, the following 1969 flight intervals can be defined to verify reliability:

| | |
|---|---|
| Mission Phase 1: (0.3 hour) | For the period from liftoff through boost and the accomplishment of spacecraft injection. |
| Mission Phase 2: (4280 hours) | For the period after spacecraft injection through cruise (including midcourse corrections) and the accomplishment of simulated capsule separation. |

From analyses made of essentially identical subsystems in the 1971 spacecraft, the 1969 flight reliability projected for these mission phases is as follows:

| Phases | Reliability |
|---|---|
| 1 only | 0.983 |
| 2 only | 0.880 |
| 1 and 2 cumulative | 0.865 |

## 5. WEIGHT AND MASS PROPERTIES

The following paragraphs discuss the weights for the Voyager 1969 test spacecraft. Also included are weight and mass properties summaries. Weights for individual components are given in IV. 6 where they are listed along with component power and temperature design parameters.

The results presented indicate that weight is available for the 1969 mission to allow a basic spacecraft with a high degree of similarity to the 1971 system. It is also possible to include various test options and a substantial science payload.

### 5.1 Spacecraft Weights Summary

As discussed in IV. 1, the allowable weight for the 1969 spacecraft depends on the particular trajectory selected and on the Atlas-Centaur performance. Selection of the trajectory in turn depends on the science payload and its objectives along with spacecraft test objectives and schedule constraints. For purposes of the present discussion a conservative reference value of 1400 pounds has been chosen for the separated spacecraft weight. This allows a basic spacecraft configuration that achieves a high degree of similarity with the 1971 design and at the same time provides an allowance for spacecraft science payload and test options. Additional payload weight of as much as 300 pounds or more can be made available depending on tradeoffs in trajectory parameters and schedule considerations.

Table 4-2 summarizes the weights for the 1969 spacecraft. The 1971 vehicle weights are shown in the same table for comparison. The weight summary includes both a weight margin and contingency. The weight margin is the difference between the reference allowable space-

Table 4-2. Comparison of 1969 Test Vehicle with 1971 Vehicle

| Item | Weight, lb. 1971 | Weight, lb. 1969 |
|---|---|---|
| Spacecraft Bus | | |
| Mechanical and Pyrotechnics | 37 | 25 |
| Spacecraft Structure | 489 | 291 |
| Thermal Control | 50 | 20 |
| Telecommunications | 160 | 136 |
| Electrical Power | 314 | 224 |
| Electrical Distribution | 142 | 67 |
| Central Sequencing and Command | 27 | 27 |
| Stabilization and Control | 100 | 72 |
| Science Support | 114 | -- |
| Margin | 187 | 187 |
| Contingency | 113 | 84 |
| Spacecraft Propulsion System | | |
| Retropropulsion | 336 | -- |
| Midcourse Propulsion | | |
| Inert Weight | 75 | 49 |
| Midcourse Propellant Unused | 215 | 38 |
| Evasive Maneuver Propulsion | 2 | 2 |
| Contingency | 29 | 5 |
| Spacecraft Science Payload and Optional Allowance | 267 | 166 |
| Spacecraft Weight in Orbit | 2,657 | |
| Propulsion | | |
| Retropropellant for Deboost | 2,733 | -- |
| Inerts Expended | 70 | -- |
| Spacecraft Weight After Capsule Separation | 5,460 | -- |
| Flight Capsule | 2,300 | -- |
| Spacecraft Weight After Midcourse Correction | 7,760 | 1,393 |
| Propulsion | | |
| Median Midcourse Propellant Used | 40 | 7 |
| Separated Planetary Vehicle | 7,800 | 1,400 |
| Adapter Weight Above Field Joint Remaining with Centaur | 12 | 54 |
| Adapter Allocated Weight Not Used | 238 | -- |
| Total Planetary Vehicle Weight | 8,050 | 1,454 |

craft weight and the design weight. This margin may be used for additional redundancy, scientific experiments, or optional equipment. The contingency allows for uncertainties in weight estimation techniques, slight modification of the design, and for balance weights to maintain the desired center of mass location. It also includes an allowance for weight growth during design completion and the development phase of the spacecraft.

In the present discussion the 1969 weight margin has been taken as 187 pounds, which is the same as for the 1971 system.

The remainder of the margin, corresponding to the reference 1400 pounds of allowable spacecraft weight, has been allocated as a design weight for science payload and test options. This amounts to 166 pounds as compared with 267 pounds for the 1971 spacecraft. A weight contingency of 6 per cent has been included to reflect the over-all level of confidence of the weight estimates at the current level of design.

### 5.2 Subsystem Weights

The weights, where possible, are the same as those for the 1971 mission. Major differences between the 1969 and 1971 missions are the retropropulsion and science subsystems. A discussion of the weights for each subsystem follows.

#### 5.2.1 Mechanical and Pyrotechnics

The 1969 test vehicle has a launch vehicle separation system similar to the one for the 1971 spacecraft except it contains four bolts instead of three. A capsule separation is simulated and is identical to that used in the 1971 spacecraft.

#### 5.2.2 Spacecraft Structure

The structure is divided into the following four parts:

1) Meteoroid protection panels
2) Framework
3) Equipment mounting provisions
4) Miscellaneous mounts

The meteoroid protection panels which encompass the bus external surface consists of 1.0-inch-thick core (3.1 $lb/ft^3$) sandwiched between two 0.025-inch-thick aluminum faces, two 0.04 $lb/ft^2$ bond lines, and 0.04-inch-thick aluminum closing channels. Detailed weights were calculated from this information.

The framework consists of various aluminum frames used for carrying the primary loads and for attaching the meteoroid protection panels. Cross sectional areas were determined by stress analysis. Equipment mounting is provided on four panels by two channels, one I-beam and two hat section beams, metal inserts in the honeycomb, and a cradle for the attitude control system.

### 5.2.3 Thermal Control

Thermal control consists of insulation, louvers, heaters, and thermostats. Twenty sheets of aluminized mylar cover all bus external surfaces. Detailed insulation weight calculations were made using this information.

The louver system utilizes the Pioneer bimetal actuator and the Mariner type louver. This combination weighs 0.56 $lb/ft^2$ and covers 10.2 square feet of the spacecraft.

Heaters and thermostats were assumed to weigh 1 pound

### 5.2.4 Communications and Data Handling

The communications and data handling subsystem weights are identical to the 1971 vehicle as shown in Table 3, Section IV, Volume 4 except that the VHF antenna, medium gain antenna, and associated equipment have been removed.

### 5.2.5 Electrical Power

The solar array weights are based on 100 square feet of solar cell area. The solar paddle structure consists of a 1-inch thick core (1.6 $lb/ft^3$) sandwiched between two 0.010-inch-thick aluminum faces, two 0.02 $lb/ft^2$ bond lines, and 0.02-inch-thick aluminum closing channels. All other electrical power components are the same as those on the 1971 spacecraft (Table 4a, Section IV, Volume 4).

### 5.2.6 Electrical Distribution

Cabling and connector weights are based on empirical data considering the amount of equipment requiring power and electrical connection, the spacecraft geometry, and the packaging technique used. Three J-boxes are used at an estimated weight of 5 pounds each.

### 5.2.7 Central Sequencing and Command

This subsystem is identical to its 1971 counterpart.

### 5.2.8 Stabilization and Control

The stabilization and control subsystem contains the same components as the 1971 Voyager with the identical valving and regulation but smaller tankage, modified plumbing, and lower thrust nozzles.

### 5.2.9 Midcourse Propulsion Systems

The monopropellant midcourse correction system is identical to its 1971 counterpart except that a single tank, off-loaded, is used instead of two tanks and the plumbing routing is modified.

The total midcourse propellant (45 pounds) is based on 75 meters/sec the same correction capability as was used to size the 1971 system.

## 5.3 Weights for Optional Equipment

Depending on over-all program objectives and schedule constraints, various test options and science payloads will probably be selected on the 1969 mission. These include such things as the planet-oriented package, retropropulsion igniter, liquid injection thrust vector control (LITVC), planetary probe, probe-spacecraft radio link and various individual science experiments. Weights for the spacecraft elements are shown in Table 4-3. The weights for the 1971 science payload are also given for reference in Table 4-4. The inclusion of the DAE is especially attractive to test the spacecraft-DAE interface. The options of the POP and a small entry capsule are mutually exclusive because of space limitations.

## 5.4 Moments of Inertia and Center of Mass

Centroidal moments of inertia have been determined computationally for the flight configuration of the spacecraft. Table 4-5 lists the moments

Table 4-3. Optional Equipment

| Item | No. of Items | Weight, Pounds |
|---|---|---|
| Retropropulsion Elements | | (50.0) |
| LITVC System | | 45.0 |
| Igniter | | 5.0 |
| Capsule-Spacecraft Link | | (9.7) |
| VHF Antenna | | 4.5 |
| VHF Receiver | 2 | 4.0 |
| VHF Preamp | 2 | 0.4 |
| Capsule Demodulator | 2 | 0.8 |
| Planet-Oriented Package Support | | (103.8) |
| POP Science Package Structure | | 41.9 |
| POP Support Shaft | | 6.2 |
| POP Support Fork | | 11.6 |
| POP Thermal Control | | 2.5 |
| Fork Bearing Housing | 2 | 4.0 |
| Drives | 2 | 10.4 |
| Cable Wrapups | 2 | 6.5 |
| Two Position Pickoffs | 2 | 5.0 |
| Science Cabling and Connectors | | 10.0 |
| Attachments and Miscellaneous | | 5.7 |

Table 4-4. 1971 Voyager Spacecraft Science Payload

| Component | No. of Items | Total Weight, Pounds |
|---|---|---|
| POP Mounted | | |
| TV Experiment | 1 | 36 |
| UV Spectrometer | 1 | 18 |
| Scan Radiometer | 1 | 10 |
| IR Spectrometer | 1 | 20 |
| Meteoroid Flash | 1 | 5 |
| Mars Sensor | 1 | 12 |
| Bus Mounted Sensors | | |
| Meteoroid Impact | 4 | 6 |
| Magnetometers | 2 | 2.6 |
| Plasma | 2 | 4 |
| Cosmic Ray | 4 | 3 |
| Trapped Radiation | 3 | 9 |
| Ionosphere Experiment | 1 | 3 |
| Bus Mounted Remote Hardware | | |
| TV Experiment | 1 | 16 |
| UV Spectrometer | 1 | 7 |
| Scan Radiometer | 1 | 2 |
| IR Spectrometer | 1 | 2 |
| Meteoroid Flash | 1 | 5 |
| Meteoroid Impact | 4 | 10 |
| Magnetometers | 2 | 10 |
| Plasma | 2 | 5.5 |
| Cosmic Ray | 4 | 5.0 |
| Trapped Radiation | 1 | 12.5 |
| Ionosphere Experiment | 1 | 6.0 |
| Data Automation Equipment | | 57 |
| TOTAL | | 266.6 |

Table 4-5. Voyager 1969 Test Vehicle

| Condition | Weight, Pounds | Center of Mass* Inches | Moments of Inertia Slug-Ft$^2$ | | |
|---|---|---|---|---|---|
| | | Station | $I_{Pitch}$ | $I_{Yaw}$ | $I_{Roll}$ |
| Separated Spacecraft Weight | 1400 | 79.8 | 221 | 268 | 374 |

*Measured from Station 0

of inertia about the pitch, yaw and roll axes as defined in Figure 4-2. Also included in Table 4-5 are longitudinal center of mass values which are measured from the aft end of the interstage structure at Atlas Centaur station 156.45 and spacecraft station 0.0. The center of mass was calculated using weights listed in Table 4-2 and with components located as shown in Figure 4-2.

## 6. COMPONENT DESIGN PARAMETERS

Design parameters for the 1969 Voyager test spacecraft are listed in Table 4-6.

Table 4-6. Component Design Parameters Voyager 1969 Test Vehicle

| | No. of Items | Total Weight (lbs) | Volume Each (in$^3$) | Electrical Power and Sources | | | Allowable Operating Temperature $^oF$ | | Allowable Nonoperating Temperature $^oF$ | |
|---|---|---|---|---|---|---|---|---|---|---|
| | | | | Average watts | Peak watts | Primary Power Source | min | max | min | max |
| Mechanical and Pyrotechnics | | 24.8 | | | | | | | | |
| Launch Vehicle Separation | | 6.6 | | | | | -65 | 165 | -300 | 165 |
| Capsule Jettison | | 7.6 | | | | | -300 | 165 | -300 | 165 |
| Solar Array Mechanism | | 9.2 | | | | | -250 | 240 | -250 | 240 |
| Attachment and Miscellaneous | | 1.4 | | | | | -250 | 240 | -250 | 240 |
| Spacecraft Structure | | 291.0 | | | | | | | | |
| Meteoroid Protection Panels | 4 | 157.5 | | | | | -250 | 240 | -250 | 240 |
| Framework | | 15.8 | | | | | -250 | 240 | -250 | 240 |
| Equipment Mounting Provisions | | 92.0 | | | | | -250 | 240 | -250 | 240 |
| High Gain Antenna Supports | | 6.8 | | | | | -250 | 250 | -250 | 240 |
| Low Gain Antenna Supports | 2 | 1.6 | | | | | -250 | 240 | -250 | 240 |
| Stabilization and Control Supports | | 0.8 | | | | | -250 | 240 | -250 | 240 |
| Attachment and Miscellaneous | | 16.5 | | | | | -250 | 240 | -250 | 240 |
| Thermal Control | | 20.2 | | | | | | | | |
| Insulation | | | | | | | | | | |
| Aluminized Mylar | | 10.3 | | | | | -300 | 300 | -300 | 300 |
| Refrasil Batt | | 3.2 | | | | | -300 | 2000 | -300 | 2000 |
| Louvers | | 5.7 | | | | | -100 | 250 | -100 | 250 |
| Heaters and Thermostats | | 1.0 | | 2.5 | 5.0 | 50 vdc | 40 | 90 | NA | NA |
| Telecommunications | | 135.8 | | | | | | | | |
| Mod-Exciter | 2 | 6.0 | 90 | 2 | 2 | 4.1 kc | 30 | 110 | -25 | 175 |
| Four Port Hybrid Ring | 1 | 0.6 | 15 | | | | 30 | 110 | -25 | 175 |
| Low Power Transmitter | 1 | 3.5 | 105 | 10 | 10 | 4.1 kc | 30 | 110 | -25 | 175 |
| Power Amplifiers (20w) | | | | | | | | | | |
| Tube | 2 | 4.0 | 84 | | | | 30 | 185 | -25 | 250 |
| Power Supply | 2 | 11.0 | 135 | 90 | 90 | 50 vdc | 30 | 110 | -25 | 175 |
| Transmitter Selector | 1 | 0.8 | 15 | 0.8 | 0.8 | 4.1 kc | 30 | 110 | -25 | 175 |
| S-band Receiver | 3 | 15.0 | 150 | 7.5 | 7.5 | 4.1 kc | 30 | 110 | -25 | 175 |
| Receiver Selector | 1 | 0.8 | 15 | 0.8 | 0.8 | 4.1 kc | 30 | 110 | -25 | 175 |
| Command Demodulator | 2 | 4.0 | 30 | 1.5 | 1.5 | 4.1 kc | 30 | 110 | -25 | 175 |
| Three Port Circulator Switch | 4 | 7.3 | 23 | | | | 30 | 110 | -25 | 175 |
| Diplexer | 3 | 2.4 | 46 | | | | 30 | 110 | -25 | 175 |
| DTU | 2 | 6.0 | 75 | 4.0 | 4.0 | 4.1 kc | 30 | 110 | -25 | 175 |
| Signal Conditioner | 1 | 1.0 | 20 | 1 | 1 | 4.1 kc | 30 | 110 | -25 | 175 |
| Data Storage | 1 | 4.0 | 100 | 1.5 | 1.5 | 4.1 kc | 30 | 110 | -25 | 175 |
| Bulk Storage | 2 | 24.0 | 350 | 5 | 15 | 4.1 kc | 30 | 110 | -25 | 175 |
| Low Gain Antenna | 2 | 2.0 | | | | | | | | |
| High Gain Antenna | 1 | 43.4 | | 1.6/1.0 | 40/ 33.0 | 4.1 kc/ 40 cps | -350 | 360 | -350 | 360 |
| Electrical Power | | 224.1 | | | | | | | | |
| Solar Array* | 1 | 100.0 | | | | | -184 | 248 | -184 | 248* |
| Batteries | 2 | 80.4 | 720 | | | | 50 | 90 | 50 | 90 |
| Inverter 250w 4.1 kc | 2 | 7.0 | 72 | | | | -20 | 120 | -50 | 200 |
| Inverter 30w 820 cy | 2 | 4.0 | 36 | | | | -20 | 120 | -50 | 200 |
| Inverter 50w 410 cy | 2 | 6.0 | 48 | | | | -20 | 120 | -50 | 200 |
| Battery Regulator | 2 | 10.4 | 192 | | | | -20 | 120 | -50 | 200 |
| Power Control Unit | 1 | 6.3 | 180 | | | | -20 | 120 | -50 | 200 |
| Shunt Element Assembly | 1 | 10.0 | 216 | | | | -20 | 120 | -50 | 200 |

*100 ft$^2$ array

Table 4-6. Component Design Parameters Voyager 1969 Test Vehicle (Continued)

| Component | No. of Items | Total Weight (lbs) | Volume Each (in$^3$) | Electrical Power and Sources: Average watts | Electrical Power and Sources: Peak watts | Electrical Power and Sources: Primary Power | Allowable Operating Temperature °F: min | Allowable Operating Temperature °F: max | Allowable Nonoperating Temperature °F: min | Allowable Nonoperating Temperature °F: max |
|---|---|---|---|---|---|---|---|---|---|---|
| Electrical Distribution | | 67 | | | | | | | | |
| Cabling Connectors | | 50 | | | | | | | | |
| J-Boxes | 3 | 15 | 216 | | | | | | | |
| Umbilicals | 2 | 2 | | | | | | | | |
| Central Sequencing and Command | | 26.6 | | | | | | | | |
| Input Decoder | 2 | 2.0 | 20 | 3.6 | 3.6 | 4.1 kc | -31 | 167 | -31 | 167 |
| Command Decoder | 2 | 2.0 | 20 | 1.0 | 1.0 | 4.1 kc | -31 | 167 | -31 | 167 |
| Sequencer | 2 | 15.0 | 200 | 9.3 | 9.3 | 4.1 kc | -31 | 167 | -31 | 167 |
| Power Supply | 2 | 7.6 | 90 | 4.1 | 4.1 | 4.1 kc | -20 | 120 | -50 | 200 |
| Stabilization and Control | | 72.0 | | | | | | | | |
| Control Electronics Assembly | 1 | 13.0 | 216 | 9 | 45 | 4.1 kc | 30 | 130 | | |
| Gyros and Electronics | 1 | 10.0 | 180 | 6/1.5/10 | 27/1.5 10 | 80 cy/ 4.1 kc dc | 30 | 130 | | |
| Coarse Sun Sensor | 4 | 2.0 | 2 | | | | 30 | 140 | -20 | 160 |
| Coarse Sun Sensor | 4 | 2.0 | 2 | | | | | | | |
| Fine Sun Sensor | 1 | 0.6 | 32 | | | | 30 | 130 | -20 | 160 |
| Canopus Sensor + Electronics | 2 | 11.0 | 220 | 3.0 | 3.0 | 4.1 kc | -30 | 100 | -30 | 100 |
| Gas Vessel + Transducers | 2 | 14.0 | 1150 | | | | 40 | 140 | 0 | 200 |
| $N_2$ Gas | | 10.0 | | | | | 40 | 140 | 0 | 200 |
| Pressure Regulator + Transducer | 2 | 3.0 | 21 | | | | 40 | 140 | 0 | 200 |
| Valves + Plumbing Set | 2 | 8.0 | | 0 | 48.0 | 50 vdc | | | | |
| Earth Detector | 1 | 0.3 | 7 | 0.2 | 0.2 | 4.1 kc | 30 | 130 | -20 | 160 |
| Propulsion System | | 49.4 | | | | | | | | |
| LITVC Simulation | | 2.0 | | | | | 40 | 90 | 30 | 100 |
| Midcourse Propulsion | | | | | | | | | | |
| Containers | | 17.0 | | | | | | | | |
| Pressurization | | | | | | | | | | |
| Hand Valve | | 0.4 | | | | | 40 | 90 | 35 | 125 |
| Lines | | 0.2 | | | | | 40 | 90 | 35 | 125 |
| Fittings and Clips | | 0.1 | | | | | 40 | 90 | 35 | 125 |
| Propellant System | | | | | | | | | | |
| Hand Valve | | 1.0 | | | | | 40 | 90 | 35 | 125 |
| Lines | | 1.9 | | | | | 40 | 90 | 35 | 125 |
| Fitting and Clips | | 0.5 | | | | | 40 | 90 | 35 | 125 |
| Bladder System | | 4.8 | | | | | 40 | 90 | 35 | 125 |
| Thrust Chamber and Valves | | 5.0 | | | | | 40 | 90 | 35 | 125 |
| Propulsion Module Structure | | | | | | | | | | |
| Container Supports | | 12.8 | | | | | 40 | 90 | 35 | 125 |
| Thrust Structure | | 2.3 | | | | | 40 | 90 | 35 | 125 |
| Attachment and Miscellaneous | | 0.9 | | | | | 40 | 90 | 35 | 125 |
| Thermal Control | | 1.0 | | | | | 40 | 90 | 35 | 125 |
| Pressurant | | 1.2 | | | | | 40 | 90 | 35 | 125 |
| Unused Propellants | | 4.0 | | | | | 40 | 90 | 35 | 125 |
| Jet Vane Assembly | | 0.2 | | | | | 40 | 90 | 35 | 125 |
| Jet Vane Actuators | | 4.0 | | 6.0 | 12.0 | 50 vdc | 40 | 90 | 35 | 125 |
| Evasive Maneuver Propulsion | | 2.0 | | | | | 40 | 90 | 35 | 125 |

# V. SUBSYSTEM DESCRIPTIONS

## 1. TELECOMMUNICATIONS

The telecommunications subsystem, which includes communications and data handling, supports the over-all spacecraft test. In addition, the 1969 test flight will provide evaluation of the telecommunications elements and the associated operations procedures, thereby enhancing the probability of success for the 1971 mission. Extensive diagnostic telemetry will be used in order to obtain a maximum of engineering data so as to reduce interpretive ambiguities to a minimum.

### 1.1 Configuration

The telecommunications subsystem has the same configuration for the 1969 flight test spacecraft as that for 1971 with the exception that (1) the capsule link receiving system (VHF antenna, receiver, demodulator) is not included, and (2) the medium-gain antenna with its associated gimbal drive system is replaced by a second fixed low-gain antenna at the forward end of the spacecraft.

The 1969 telecommunications electronic equipment is installed on spacecraft panel II of Figure 4-5 in an identical manner to the installation of the same equipment on panel VI for the 1971 spacecraft, as shown in Figure 7 of VS-4-550. A list of the 1969 telecommunications equipment with associated design parameters is given in Section IV.6. Volume 2 gives a functional description of the communications subsystem in VS-4-310 and the data handling subsystem in VS-4-311. The tape recorder is described in VS-4-312.

### 1.2 Antennas

The 1969 high-gain antenna design and mission usage is identical to the 1971 mission. It is stowed in a different way during powered flight as shown in Figure 4-2 and 4-3. Once deployed, the gimballing capability is the same as for the 1971 configuration and utilizes an identical drive mechanization. The 1969 aft low-gain antenna is identical to that for the 1971 configuration and is installed with the same cone angles for its axes as shown in Figure 4-4. The second (forward) low gain antenna is oriented with its principal axis at a 180-degree cone angle as shown in Figure 4-4. The location and orientation of this antenna, in conjunction with the aft low gain antenna, provides full spherical coverage.

## 1.3 Data Handling

The complete 1971 data handling capability will be provided on the 1969 spacecraft. Because of its importance, the DAE-data handling interface should be evaluated to the maximum extent possible during the 1969 test flight. Ideally, the actual DAE would be flown with on-board simulation of science data. In particular, the high rate science data should be simulated so as to permit tape recorder evaluation. In this regard, it is planned to simulate the encounter mode high rate science with a known code sequence formatted into words and frames and read into the tape recorders at the normal 163,812 bits/sec. PN generators could be utilized to generate the sequence.

## 1.4 DSIF Interface

The DSIF interface has both operational and functional aspects. The 1969 mission will provide testing of procedures, spacecraft control, and software functions of data handling and reduction. In the functional interface the mission will provide checkout of mission-peculiar DSIF station equipment, and compatibility testing of the spacecraft transponder for command, telemetry, turn-around ranging, and doppler.

## 1.5 Capsule Link

For a Mars flyby 1969 mission which does not include a capsule, there does not appear to be justification for including capsule receiving equipment on board the spacecraft. Earth-based VHF transmission to the spacecraft is feasible during the early portion of the flight but this would not provide simulation of the entry sequence or of the multipath propagation.

In the event that a capsule is included for the 1969 mission, the VHF receiving equipment and antenna will be included on the spacecraft.

## 1.6 Spacecraft Integration

The test flight will afford a relatively early opportunity to integrate an entire complement of subsystems, thereby establishing interface compatibility. RFI data can be obtained on equipment susceptibility and on radiated interference spectra. In addition to its application to spacecraft equipment integration, this will be of value in defining the science equipment interface for the 1971 mission.

## 2. STABILIZATION AND CONTROL

The stabilization and control subsystem (SCS) for the Voyager 1969 flight test spacecraft must establish and maintain three-axis stabilization and control in the same manner as for the 1971 mission, with the exception of the retropropulsion maneuver. While the function of the SCS is to support the over-all 1969 mission, it is in keeping with this mission to incorporate engineering objectives related to the 1971 system. The techniques used to estimate the disturbance torques acting on the spacecraft will be verified by the telemetered signals from the switching amplifiers which drive the reaction control jets. Checks will be made on procedures for the 1971 mission, such as verification of Canopus acquisition and verification of maneuver accuracy by prepositioning the high gain antenna. The 1969 mission will also provide an opportunity to verify the procedures for integration and checkout of the SCS and support equipment.

### 2.1 Configuration

The 1969 SCS will be comprised of components identical to the 1971 SCS with the exception of the reaction control system and will operate in the same way. This results from the fact that the two spacecraft system requirements include nearly all of the same functions.

The primary attitude references are the sun and the star Canopus as for the 1971 mission. The layout of the 1969 spacecraft shows that the optical sensors for the 1971 spacecraft can be incorporated readily into the 1969 SCS. The simple and compact design of the sun sensor elements enables their installation with only minor modifications to the cabling. The Canopus sensor will be integrated into the 1969 vehicle with an adequate, unobstructed field of view.

Geometry for the location and orientation of the SCS sensors and reaction jets is shown in Figure 4-4. The tankage is installed on the spacecraft midcourse propulsion module in the same way as for the 1971 configuration. The electronics is installed on spacecraft panel I of Figure 4-5 in an identical manner to the installation of the same equipment on panel III for the 1971 spacecraft as shown in Figure 4 of VS-4-550.

The control electronics assembly input-output characteristics will be identical for both SCS designs with the exception of the omission of the LITVC from the 1969 vehicle. Dummy loads will be provided so that the 1971 characteristics for the LITVC actuators are duplicated in the 1969 test.

A list of the 1969 SCS equipment with associated design parameters is given in Section IV.6. A functional description of the SCS is given in Volume 2, VS-4-410.

## 2.2 Reaction Control System

The reaction control system will consist of independent gas supplies, valves, and jets as for the1971 design. Both normal level jets and high level jets will be included, the latter only for test evaluation as they are normally used only during the retropropulsion maneuver. The differences in vehicle reaction control requirements will be accommodated by changes in some of the passive components. Present estimates of thrust levels show that the flow requirements of both systems can be satisfied by the same regulators and valves. Thus the dynamic components of the reaction control system will be identical for both the 1969 and 1971 designs.

The gas vessels for 1969 will be smaller than for 1971 but the design, materials, and fabrication processes will be identical. The nozzles for the1969 design will provide lower thrust than those for 1971 but, again, the same design, materials, and processes will be used. The plumbing will be different because of the differences in spacecraft layout. In addition, it will be necessary to use either flexible or coiled tubing to carry the pneumatic lines across the solar panel hinges. The reaction control heater elements will be the same for 1969 and 1971.

The asymmetric 1969 configuration does not allow the use of coupled pairs of jets for reaction control for all axes, and pitch control torques will be provided by unbalanced forces. The effect of this mechanization on trajectory dispersions will be negligible since the sign of the applied force alternates during limit cycle operation and, therefore, the linear impulse imparted to the spacecraft tends to have zero average value. The net velocity imparted to the spacecraft during reorientation maneuvers is negligible because of the low thrust levels. Roll and yaw control torques will be provided by jet pairs acting as a force couple.

### 2.3 Performance Parameters

As in the 1971 system, a cruise control deadband of $\pm$ 0.5 degree will be combined with a limit cycle rate of 6 degrees per hour to provide economical cruise operation. The fine control deadband of $\pm$ 0.25 degree will be available on command from the central sequencing and command subsystem as it is for the 1971 SCS. The control acceleration will be about 1.2 $mr/sec^2$ about each of the control axes.

The jet vane actuator parameters for the 1971 design are entirely compatible with the 1969 vehicle.

### 2.4 SCS Analysis

#### 2.4.1 Spacecraft Model

The model of the 1969 spacecraft used for preliminary design of the SCS is given below.

| Axis | Moment of Inertia | Reaction Control Moment Arm | Center of Gravity |
|---|---|---|---|
| Pitch | 221 slug-$ft^2$ | 11.2 ft | 0 |
| Yaw | 268 slug-$ft^2$ | 11.1 ft | 1.4 inches |
| Roll | 374 slug-$ft^2$ | 11.1 ft | 70 inches |

Spacecraft weight = 1400 pounds

#### 2.4.2 Jet Thrust Level

The jet thrust level has been selected to provide a compromise between requirements for cruise gas economy and control of disturbance torques. The cruise limit cycle rate has been selected so that gas consumption is not significantly affected by the expected solar radiation pressure torques.

The jet thrust level computation is based on a 6.0 degree per hour limit cycle rate ($\theta_{LC}$) and a 50-millisecond firing time ($t_{on}$) during cruise.

The standard relationship is

$$\dot{\theta}_{LC} = \frac{\ddot{\theta} \times t_{on}}{2}$$

Therefore $\ddot{\theta}$, the corresponding angular acceleration is given as

$$\ddot{\theta} = 1.17 \times 10^{-3} \text{ rad/sec}^2$$

The control torque $T_o$ based on the maximum moment of inertia is given as

$$T_o = I_{max} \times \ddot{\theta} = 0.436 \text{ ft-lb}$$

The thrust level for each reaction control jet is then

$$\text{Thrust} = \frac{0.436 \text{ ft-lb}}{22.2 \text{ ft}} = 0.0196 \text{ lb}$$

### 2.4.3 Cruise Mode Duty Cycle Characteristics

The duty cycle characteristics for the cruise mode are based on a ± 0.5-degree limit cycle amplitude and a 200-day transit. Thus

$$\frac{\text{pulses}}{\text{jet}} = 6 \frac{\text{pulses}}{\text{hr}} \times 24 \frac{\text{hr}}{\text{day}} \times 200 \frac{\text{days}}{\text{transit}} = 28,800$$

For two jets per axis and three control axes, the total number of pulses is 173,000. A firing time of 50 milliseconds per pulse gives a total cruise firing time of 8650 seconds. The fuel consumption for cruise is based on cold $N_2$ with an $I_{sp}$ of 60. Therefore

$$\text{Fuel consumption} = \frac{0.0196 \text{ lb} \times 8650 \text{ sec}}{60} = 2.82 \text{ lb}$$

### 2.4.4 Solar Disturbance Torques

Based on the configuration geometry and typical sun-spacecraft-earth angles, the solar radiation torques $T_D$ is computed as:

$$T_D = 4.15 \times 10^{-5} \text{ ft-lb at earth}$$

$$T_D = 7.1 \times 10^{-5} \text{ ft-lb at the transit midpoint}$$

$$T_D = 3.24 \times 10^{-5} \text{ ft-lb at Mars}$$

The effect of the disturbance torque is to alter the period between gas jet firings. For the cruise limit cycle rate selected above, assuming an average value of $4 \times 10^{-5}$ lb-ft, the disturbance torque increases the fuel required from 2.82 to 3.17 pounds.

### 2.4.5 Thrust Vector Control

The SCS functions the same for the thrust vector control mode as for the 1971 vehicle midcourse correction mode. A block diagram is shown in Figure 5-1.

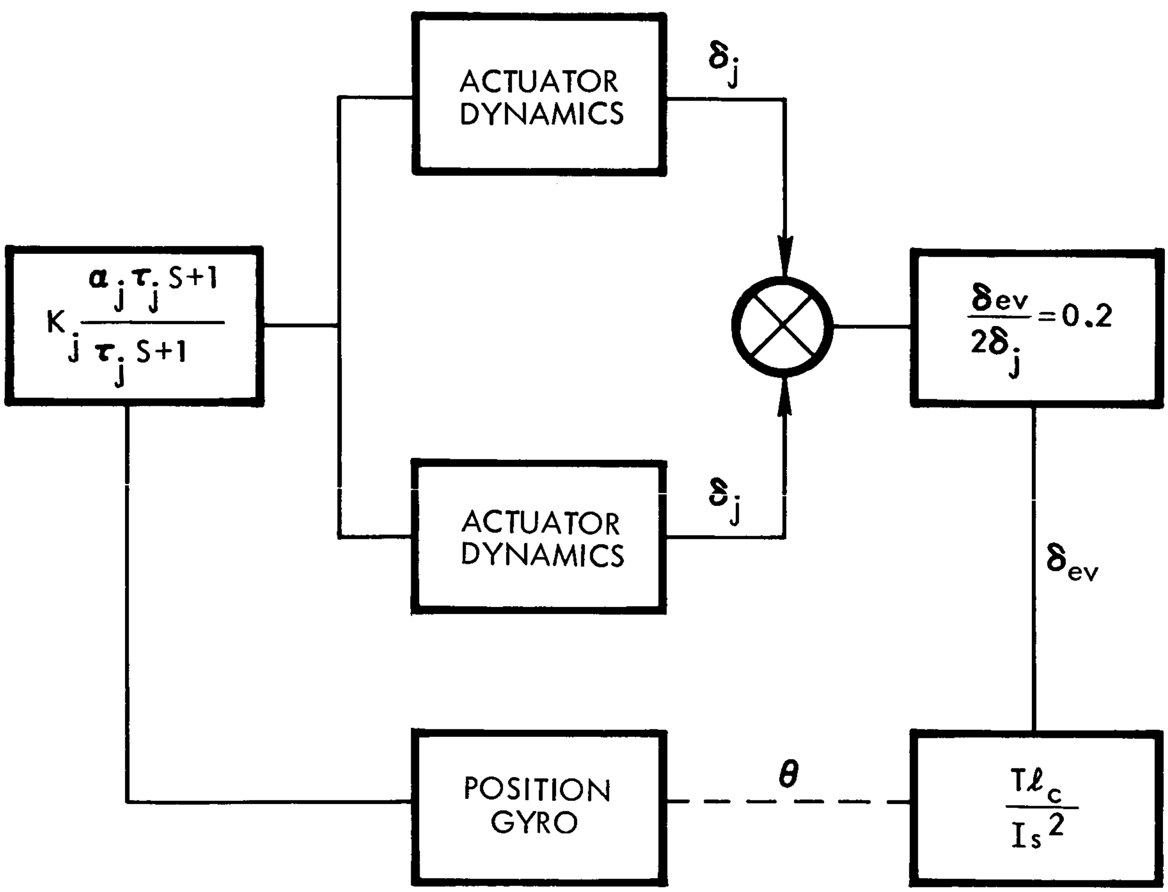
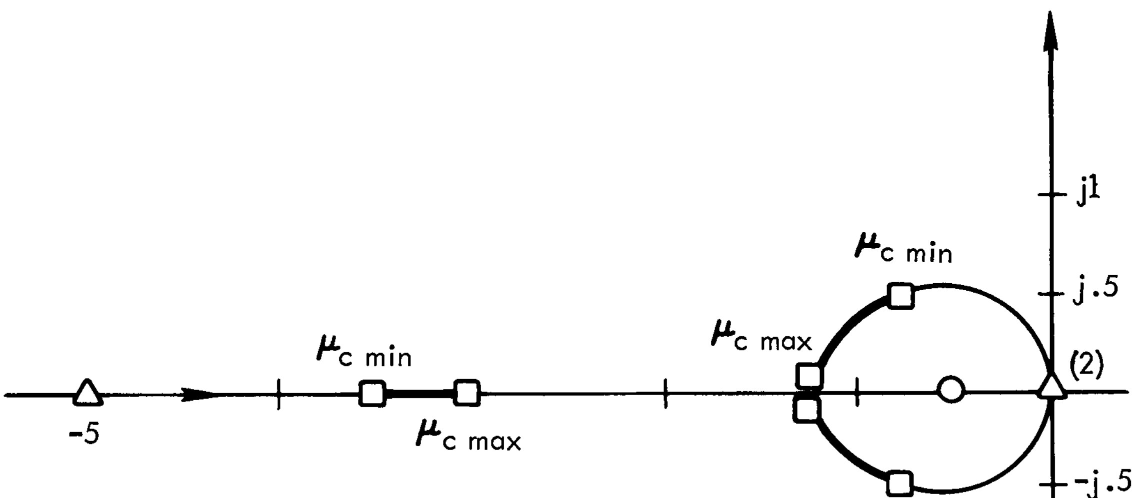



Figure 5-1. Block Diagram of Pitch or Yaw Axis Jet Vane

Based on a criterion of 1/2 maximum jet vane deflection for an 0.5-degree pointing error, the attitude error gain is 25. A lag time constant of 0.2 second and a lead-lag ratio of 10 has been selected. Figure 5-2 is a root locus plot which shows that adequate stability and response is attainable with constant SCS gains over the anticipated range of spacecraft parameters.




$$\mu_c = \frac{T\ell_c}{I}$$

$T = 50$ lb

$\mu_{c_{max}} = 0.151$

$\ell_c = 8''$

$\mu_{c_{min}} = 0.123$

Figure 5-2. Pitch and Yaw Axis Jet Vane TVC Root Locus Plot

## 3. CENTRAL SEQUENCING AND COMMAND

The 1971 central sequencing and command (CS and C) subsystem will be incorporated in the 1969 spacecraft and will operate in the same manner in the 1969 flight test as for the 1971 mission. The 1969 CS and C equipment is installed on spacecraft panel I shown in Figure 4-4 in an identical manner to the installation of the same equipment on panel III for the 1971 spacecraft, as shown in Figure 4 of VS-4-550. A list of the CS and C equipment with associated design parameters is given in Section IV.6 and a functional description is given in VS-4-450.

The basic similarity between the 1969 and 1971 systems plus the addition of simulation elements to the 1969 spacecraft will make the sequencing and command requirements essentially identical for the two missions. However, the design of the CS and C, with its centralized memory stack, makes it readily adaptable to any mission-peculiar requirements if such exist. The subsystem is essentially a black box that provides command discretes and serial pulses on output lines. The output lines can be plugged into various subsystems depending on whether they need data or not or can be sent to the telemetry system for CS and C verification.

The CS and C equipment can be exercised without the full complement of functions. The decoder, clock, and memory logic can all be tested with a nominal set of commands and data. However, exercising the basic unit with a minimum function capability does not satisfactorily test the complex interactions, mode switching, and timing problems that would be involved in the 1971 mission. A complete mission simulation is therefore important. Thus, during the 1969 mission, command and serial data outputs will be stored in dummy subsystem registers so that the processes that occur in the 1971 mission can be exercised. These registers will be sampled for telemetry and the results recorded for analysis. The registers that receive the outputs to the simulated subsystems will be the same as those utilized for the 1971 mission. The sample rates for engineering should also closely match those expected for the 1971 mission. In this way the command and telemetry loads should closely approximate the state of affairs for the 1971 mission. Problems relating to the interaction of ground commands and stored commands can then be studied, and uncertainties in sequence timing and clock ranges and granularities can be resolved. Other questions relating to the use of backup commands and on-board logic can also be approached more realistically in a real-time operation of this nature.

## 4. ELECTRICAL POWER

The 1969 flight test spacecraft provides an opportunity to obtain flight test data on the 1971 Voyager power subsystem equipment and on the over-all operation of the subsystem. Critical parameters of the power equipment will be monitored under actual flight operating conditions.

It will also be feasible to obtain performance, thermal, and degradation data on the basic solar panel design to be employed in 1971. In addition, the effect of low temperatures on array properties can be evaluated by simulation of eclipses at Mars. This may be accomplished by misorienting the spacecraft to the sun for appropriate periods.

The above considerations lead to the conclusion that the 1969 flight test will allow validation of the power subsystem and will yield a high probability of success for the 1971 mission.

### 4.1 Configuration

Except for the solar array and shunt elements assembly, the 1969 power subsystem is identical to the 1971 design and operates in the same manner. The identical elements include the power control unit, batteries, battery regulators, and power conditioning equipment.

Since there are only six electrical sections in the solar array (12 in 1971), the number of shunt elements and their associated controls are reduced to half from the 1971 solar array. However, the individual shunt elements are the same as those for 1971 and operate in a sequential manner just as for the 1971 system.

Because of the smaller booster diameter in1969, it will not be possible to flight test the fixed array designed for 1971. Instead, the 1969 solar array consists of three identical panels as shown in Figure 4-2. Also, the 1969 panels will be deployed as opposed to the fixed panels in the 1971 spacecraft. A description of the deployment mechanism is given in Section V. 12. Each panel consists of two electrical sections, as in the

1971 design. Although the panel geometry is different, the number of cells, modules, the electrical layout for each panel, and the same solar cell and basic substrate design are utilized as for the 1971 system.

The power equipment is installed on spacecraft panel IV shown in Figure 4-5. This installation is identical to that for the same equipment on panel V of the 1971 spacecraft (shown in Figure 6 of VS-4-550). A list of components and their parameters is given in Section IV. 6 and a functional description of the power subsystem is given in VS-4-460.

## 4.2 Power Capability and Loads

The total solar array area is 100 square feet as compared to 190 square feet for the 1971 spacecraft. This results in an estimated minimum array output for the 1969 spacecraft as follows:

265 watts at 1.0 AU
434 watts at 1.38 AU
265 watts at 1.67 AU

It should be noted that, since the 1969 spacecraft will not orbit Mars, the degradation due to radiation damage allowed for in the 1971 design is not applicable to the 1969 fly-by mission. Hence the 265 watts given above for 1.67 AU is based on a 20 per cent higher output at 1.67 AU than for the extended duration 1971 case.

The estimated basic loads for the 1969 spacecraft are summarized in Table 5-1. Individual equipment power levels are given in Section IV. 6. The difference between the minimum available solar array outputs and the output required ranges from 40 watts during initial cruise at 1 AU to over 200 watts at encounter. This excess power capability is available for additional loads or experiments which may be desirable for the 1969 flight test spacecraft.

In order to simulate the effects of the higher power levels of the 1971 spacecraft on the power conditioning equipment, dummy loads will be used on the 1969 test spacecraft. These dummy loads will be used to augment the basic 1969 spacecraft loads and will be resistive elements mounted external to the spacecraft.

Table 5-1. 1969 Flight Test Power Load (watts)

| Subsystem Identification | Loads | Form | Pre-launch | Launch | Acquire | Cruise | Maneuver | Encounter |
|---|---|---|---|---|---|---|---|---|
| CS&C | Electronics | 4.1 kc | 18 | 18 | 18 | 18 | 18 | 18 |
| Telecommunications | Data handling, electronics, bulk storage antenna drive electronics | | 32 | 28 | 28 | 23 | 20 | 22 |
| SCS | Cont. electronics ass'y., sensors gyro electronics | | 13 | 13 | 13 | 13 | 15 | 13 |
| Subtotal | | 4.1 kc | 63 | 59 | 59 | 54 | 53 | 53 |
| Telecommunications | 20 w TWT | 50 VDC | 0 | 0 | 90 | 90 | 90 | 90 |
| Thermal Control | Heaters | | 0 | 0 | 0 | 6 | 6 | 6 |
| SCS | Jet vane, gyro heater | | 0 | 0 | 0 | 0 | 16 | 0 |
| SCS | Reaction gas heater | | 0 | 0 | 0 | 40 | 40 | 40 |
| Subtotal | DC loads | 50 VDC | 0 | 0 | 90 | 136 | 152 | 136 |
| SCS | Gyrodrives | 820 cps | 0 | 0 | 0 | 0 | 6 | 0 |
| Telecommunications | Antenna drive | 410 cps | 0 | 0 | 1 | 1 | 1 | 0 |
| Subtotal | Drives | 820/410 cps | 0 | 0 | 1 | 1 | 7 | 0 |
| 4.1 kc inverter input | | 50 VDC | 81 | 76 | 76 | 70 | 68 | 68 |
| 820 cps inverter input | | " | 0 | 0 | 0 | 0 | 8 | 0 |
| 410 cps inverter input | | " | 1 | 1 | 2 | 2 | 2 | 1 |
| DC loads | | " | 0 | 0 | 90 | 136 | 152 | 136 |
| Total Load on DC Bus | | 50 VDC | 82 | 77 | 168 | 208 | 230 | 205 |
| Batteries + Charge Regulator* | | 50 VDC | 0 | 0 | 80 | 0 | 0 | 0 |
| Boost Regulator* | | | 20 | 20 | 6 | 6 | 58 | 6 |
| Power control unit | | 50 VDC | 10 | 10 | 10 | 10 | 10 | 10 |
| Average Solar Array Output | | | -- | -- | 264 | 224 | -- | 221 |
| Average Battery Output | | | 112 | 107 | -- | -- | 298 | -- |

*Battery charging assumes 1.6 ampere rate at 50 volts, boost regulator efficiency of 80% and no load loss of 6 watts.

## 5. ELECTRICAL DISTRIBUTION

In general, the 1969 electrical distribution equipment will be directly applicable to the 1971 Voyager despite the detailed differences in the specific hardware configurations. It is worth pointing out, however, that the 1969 flight test is not needed to validate the materials and components as such. The items tentatively planned for the 1971 Voyager electrical distribution subsystem are currently in use on orbiting spacecraft, and in general have operating histories in the space environment exceeding the lifetime requirements for the Voyager mission. However, the 1969 flight will allow evaluation of circuit designs within the electrical distribution equipment, the module and assembly packaging techniques, the intra-panel and inter-panel interconnect cabling design concepts and the cable routing plan. In particular, the use of the same spacecraft panels along with their equipment installations in 1969 and 1971 results in validation under the Centaur powered flight environment. Such a test is applicable to the 1971 system with possible limitations due to differences in the transmissibility coefficients for the basic spacecraft structure.

The 1969 electrical distribution subsystem necessarily differs from that for the 1971 Voyager spacecraft to the extent it reflects, in detail, the specific 1969 structural and external equipment configuration. Due to the differences in the basic structure configuration, the inter-panel interconnect cables will not be identical to those for the 1971 Voyager configuration. However, the basic inter-panel cabling will be implemented in a configuration that is representative of the 1971 Voyager. The lower ring harness, with breakouts to externally mounted assemblies, will be identical in concept and, as far as possible, in configuration.

The active circuitry within the electrical distribution equipment, the ordnance control circuits and the power switching circuits will be identical for the 1969 and 1971 systems, although the total number of these circuits will be different. The smaller structure should result in shorter cable runs, and the absence of the science payload, the planet-oriented package, the retropropulsion subsystem, and the flight capsule eliminates a considerable number of the electrical distribution requirements which are present in the 1971 Voyager. Certain of these, such as the planet-oriented package and the science payload, are options that may be included or simulated. Also, pyrotechnic items such as the retropropulsion igniter and liquid injection gas generator, and the capsule jettison bolts and connector, will probably be included as test installations. The ground test umbilical connector which carries the spacecraft prelaunch ground test and support interface to the Centaur will be the same as for the 1971 Voyager. The control and monitor of the safe&arm device for the retropropulsion subsystem will, of course, apply only to a test installation for the corresponding pyrotechnics. The use of such test installations allows the ordnance control circuitry to be identical with that for the 1971 Voyager; also, the ordnance devices will be the same. In some cases an ordnance device will be assigned, partially, to a different application as in the case of the solar panel deployment system.

Equipment panels containing spacecraft subsystem assemblies (shown as I, II, III in Figure 4-5) correspond to the 1971 Voyager panels, with certain minor differences as covered in the various subsystem discussions. Thus the equipment panel interconnect cabling will be essentially the same as that for the 1971 Voyager spacecraft.

The general design, fabrication, and assembly techniques for the 1969 system will be identical to those for the 1971 Voyager.

## 6. THERMAL CONTROL

### 6.1 Description

The 1969 Voyager flight test spacecraft employs the same basic thermal design concepts as for the 1971 system. The spacecraft compartment is insulated except for the louver-covered radiators and sensor apertures. Insulation, methods of insulation attachment, the louver system, equipment mounting panels, and methods of mounting electronic equipment are the same for both spacecraft. In addition, essentially the same design conditions apply except for the retropropulsion maneuver and Martian orbital phases.

The change in the structural configuration along with the absence of the solid engine and the planet-oriented packaged (POP) are the major differences between the 1969 test spacecraft and the 1971 spacecraft. Exclusion of the solid propellant engine eliminates plume heating and heat soak-back effects for the test spacecraft. Elimination of the POP reduces heat leaks and POP gimbal system electrical heater power requirements. Solar array heat leaks are reduced for the test spacecraft since array mounts are reduced from six to four. The total spacecraft power dissipation is reduced for the test spacecraft. Also, elimination of the external experiment packages and the 3-foot antenna reduces subsystem heater power requirements.

The functional description presented in Volume 2, VS-4-510 for the 1971 thermal control is applicable for the test spacecraft except as noted above. The insulation configuration for the 1969 spacecraft is indicated in Figure 4-2, and the 1969 thermal control equipment is listed along with weight and power data in IV.6.

### 6.2 Main Compartment Heat Balance

A separate compartment heat balance calculation is required for the 1969 test spacecraft. The 1969 equipment panels (shown in Figure 5-3) are essentially the same as the corresponding ones used in the 1971 system. The resulting heat balance parameters are as follows:

| | |
|---|---|
| Louver-covered radiator area | 10.2 $ft^2$ |
| Power dissipation margin over heat leaks with closed louvers for the cold condition | 70.0 watts |

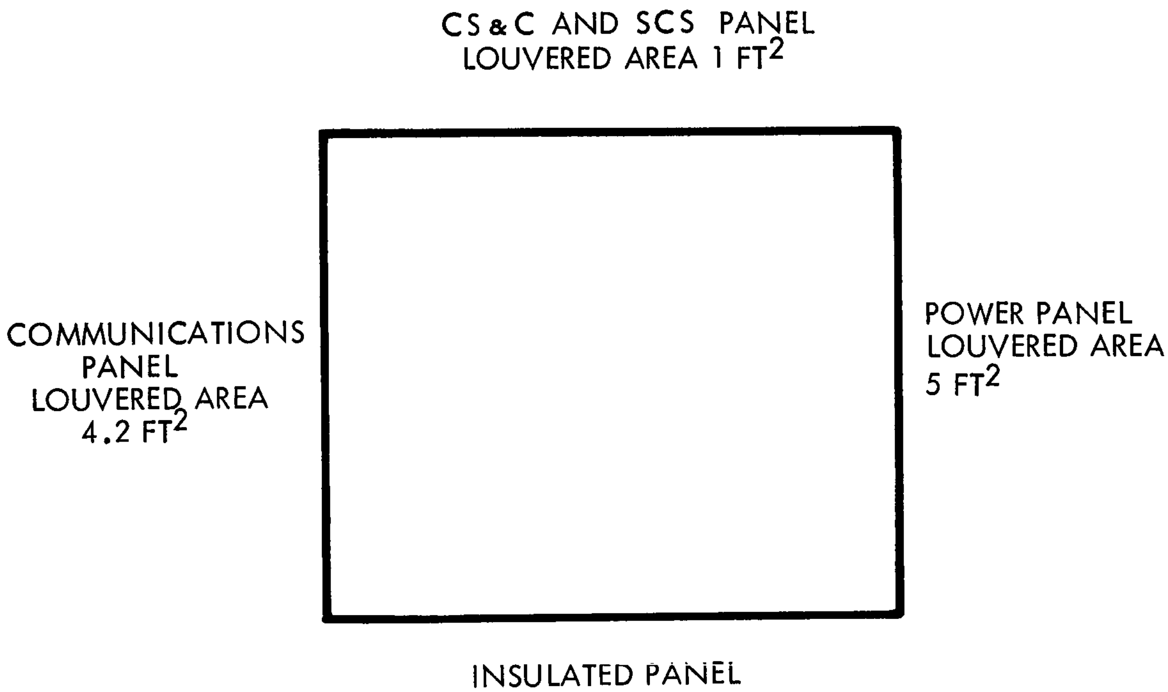



Figure 5-3. Spacecraft Thermal Control Inboard Profile

#### 6.2.1 Hot Case

a. Performance Data

The hot condition corresponds to near-earth cruise; data used for preliminary design is given below:

| Loads in Main Compartment | Watts |
|---|---|
| Electrical power subsystem (75 watts shunt dissipation) | 109.0 |
| Communications subsystem (less 20 watts RF) | 78.4 |
| Stabilization and control | 13.7 |
| Central sequencing and command | 8.5 |
| Total dissipated power, P | 209.6 |

| Heat Fluxes in Main Compartment | |
|---|---|
| Solar array attachments | 12.0 |
| Radiation from solar array | 69.0 |
| Attitude control lines | 1.0 |
| Forward insulation | -2.7 |
| Aft insulation | 0.8 |
| Side panel insulation | -18.9 |
| Net heat flux, $q_{net}$ | 61.2 |

b. Heat Balance Calculation

The net heat to be rejected by the louver system $Q_{net}$ is given by: $Q_{net} = P + q_{net} = 270.8$ watts. The heat rejection capabilities of the 10.2 $ft^2$ of louver-covered radiating area at 85°F is 306 watts. Thus the radiators have the capability of rejecting 13 percent more power than dissipated, and the average compartment temperature is less than 85°F.

6.2.2 Cold Case

The cold case corresponds to the following:

Performance Data for Encounter

o Loads in Main Compartment

| | Watts |
|---|---|
| Electrical power subsystem (25 watt shunt) | 58.0 |
| Communication subsystem (less 20 watt RF) | 78.4 |
| Stabilization and control (cruise mode) | 13.7 |
| Central sequencing and command | 8.5 |
| Total dissipated power | 158.6 |

o Heat Fluxes in Main Compartment

| | Watts |
|---|---|
| Heat loss through closed louvers (effective emissivity 0.1, panel temperature 40°F) | 31.6 |
| Heat loss through solar array attachment fittings | 8.0 |
| Heat loss through forward insulation, 22 $ft^2$ (solar constant 46.7 watts/$ft^2$; conductivity degraded 25 percent, nominally 0.05 watts/$ft^2$. | 1.1 |
| Heat loss through Canopus sensor and earth detector openings (two 5 x 5 inch and one 4 x 4 inch openings) | 14.3 |
| Heat loss through side panel insulation, 63 $ft^2$ (conductivity degraded 25 percent, nominally 0.47 watts/$ft^2$ | 29.6 |
| Heat loss from attitude control lines | 4.0 |
| Total heat loss | 88.6 |
| Margin | 70.0 |

Thus a margin of 44 percent of dissipated power is available for heat leaks and for maintaining spacecraft temperature above 40°F.

### 6.2.3 Equipment Panel Average Temperatures

Average equipment panel temperatures have not been calculated for the test spacecraft; however, they are within the louver system actuation temperature range of 40 to 85°F as demonstrated by the over-all energy balance.

## 6.3 Thermal Control Experiment

In keeping with the over-all objective, it is proposed to incorporate a thermal control experiment into the 1969 test mission. This would evaluate the degradation in thermal properties of surface coatings used on the 1971 spacecraft under solar and space exposure. Super insulated sample holders (SISH)* would be used which insulate each sample from

* Developed at TRW for the Air Force Systems Command, Research and Technology Division. Report No. RTD TDR 63-4269, "Design and Construction of Sample Holders for Orbital Temperature Control Coatings Experiment," April 1964.

the spacecraft and from the others through the use of aluminized Mylar insulation and by suspending the samples on dacron cords. Sample temperatures are measured with platinum resistance thermometers. A possible approach would be to orient one sample holder continuously toward the sun and to expose other sample holders to space but shaded from the sun to be deployed for solar irradiation at various times during transit. This would allow separate evaluation of degradation due to solar and space exposure. Each SISH weighs 0.66 pounds and carries six coating samples.

## 6.4 Relation to the 1971 Mission

The salient feature of the 1969 test flight is that it provides operational data to validate the same analytical techniques, thermal configuration data, test facilities, and simulation techniques, materials and processes, and equipment as are used for the 1971 thermal control subsystem. The similarity in the two systems allows for considerable joint applicability of the analytical and test work. Also, the 1969 flight spacecraft will be tested in the same space simulation test chamber to be used for the 1971 flight spacecraft. Both vehicles will be extensively instrumented to determine temperature differentials in areas of interest such as low conductance joints for the solar array and interstage fittings. Thus, the 1969 flight test data can be used to "calibrate" the total implementation process and to establish confidence in thermal control for the 1971 mission.

Thermal control flight test data is desired for various conditions. The earth solar eclipse portion of the injection phase serves as an indication of the performance of the thermal subsystem during a Martian eclipse. Eclipse at Mars would be desirable but not mandatory since this condition can be easily and accurately simulated during test. Transient flight test data for the anticipated worst case nonnominal attitude will serve as an important check point. When compared with data from ground test simulation of this condition, it will determine the accuracy for predictions of transient temperatures and solar energy absorbed by the louvers.

In addition to over-all checks on thermal control, the 1969 flight test will validate individual items such as the thermal design and heater power requirements for the double gimballed antenna, which is the same for the two missions. Also, a transducer to measure louver angle during flight without disturbing louver operation would be desirable. This would allow a comparison with analysis and ground test results to evaluate louver and subsystem performance. A survey for such a device will be conducted in Phase Ib.

## 7. STRUCTURE

The 1969 flight test spacecraft has a different structure from that of the 1971 spacecraft. Although it embodies many features of the 1971 vehicle, it serves essentially as the structure for the 1969 spacecraft without direct application to the 1971 design. An exception to this is the use of identical equipment mounting panels for the two spacecraft.

The primary function of the structure is to integrate, with a minimum of structural weight, the many subassemblies comprising the spacecraft. It provides sufficient strength, rigidity, and other physical characteristics necessary to maintain adequate alignment between components, acceptable static and dynamic load environments, and support spacecraft components and assemblies during preflight, boost, and cruise. Other design objectives include meteoroid protection, ease of maintenance, accessibility, and flexibility to accept changes in the mission and subsystem requirements.

### 7.1 Structual Arrangement

The structural arrangement for the 1969 spacecraft is shown in Figure 5-4. The primary structure is composed of two major components, the basic bus structure and a Centaur interstage structure. The interstage structure is located between spacecraft station 0 (Atlas-Centaur station 156.45) and spacecraft station 53, while the basic bus goes from spacecraft stations 53 to 118.5. The spacecraft-Centaur separation plane is at spacecraft station 53. The primary bus structure is in the form of a square truncated pyramid while the interstage structure is the combination of a cylinder and a truncated cone. The bus structure is composed of two major modules; the outer structure contains all the electronic subsystems while the base unit contains the midcourse propellant subsystem and the attitude control tanks. Mounting brackets for the high gain antenna are attached to the base of the bus structure. Deployable solar panels are hinged from the lower edge of the bus structure.

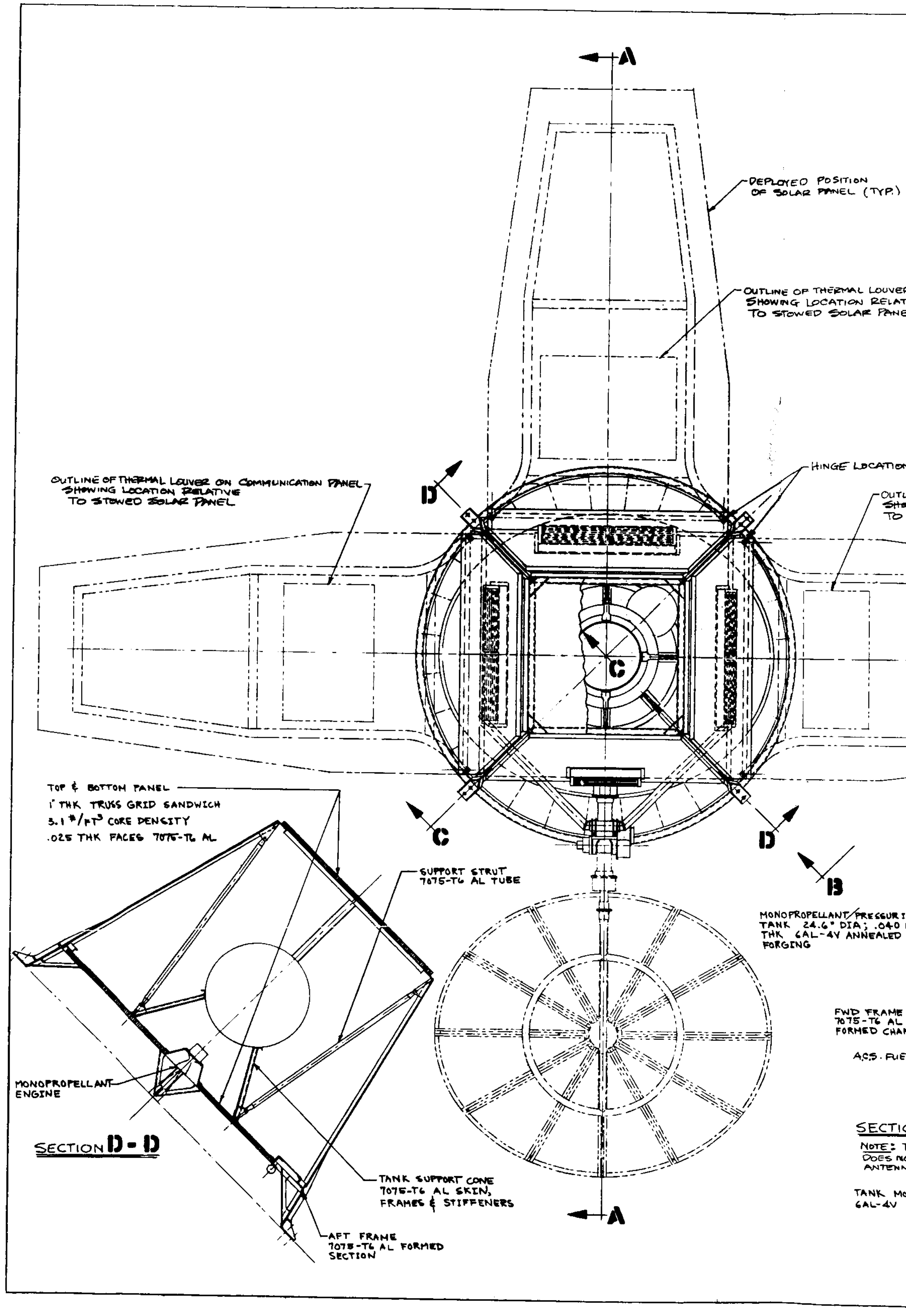
## 7.2 Bus Module Structure

The bus module is the outer structure and serves two basic functions. It is the mounting structure for all electronic subassemblies and the main load-carrying structure for the spacecraft.

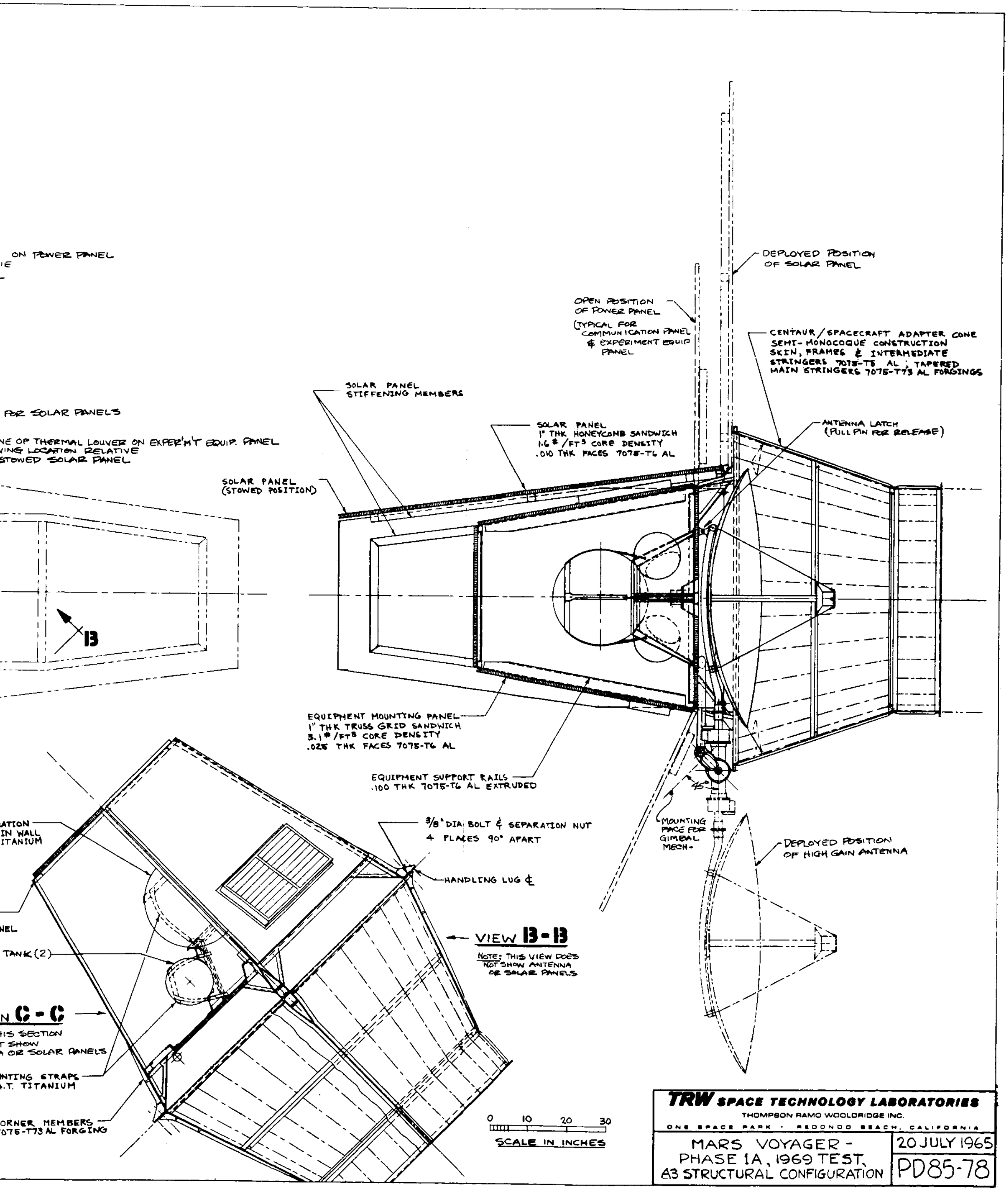
The structure consists of a four-sided frame structure with panels on the sides and top. Four machined fittings run down the corners and are the principal axial load-carrying members. They have an angle cross-section with integrally machined end fittings at the lower end for attachment to the interstage module. Material for these fittings is 7075-T73 aluminum. Attachment to the interstage structure consists of bolts with separation nuts and bolt catchers at the four fittings as described in V.11. The four fittings are joined by a channel frame at spacecraft station 117.5 and a Z-frame at spacecraft station 63. Both frames are made of 7075-T6 aluminum sheet and are gusseted to the four axial members to provide structural rigidity during fabrication and assembly.

The side panels which run from spacecraft stations 62 to 118.5 serve as mounting panels for electronic subsystems. In addition, they provide meteoroid protection, serve as a heat sink for the thermal control system, and are the main shear-carrying members of the spacecraft. The thermal louver assemblies mount to the outer face of all four panels. The panels are made of 1-inch-thick aluminum truss grid core with 0.025-inch 7075-T6 aluminum face sheets. Extruded rails are attached to the back side of the panels and serve as mounting members for all electronic equipment. Panels are bolted on all four sides to the frame and have a hinge along the lower edge of the panel to facilitate access to the equipment. The equipment mounting panels are identical in construction and method of attachment to those used on the 1971 spacecraft.

The top surface of the bus is covered with a square panel identical in construction to the side panels. It serves as a shear web and provides meteoroid protection.



A
DEPLOYED POSITION OF SOLAR PANEL (TYP.)
HINGE LOCATION
OUTLINE OF THERMAL LOUVER ON COMMUNICATION PANEL SHOWING LOCATION RELATIVE TO STOWED SOLAR PANEL
D
C
C
D
B
TOP & BOTTOM PANEL
1" THK TRUSS GRID SANDWICH
3.1 #/FT³ CORE DENSITY
.025 THK FACES 7075-T6 AL
SUPPORT STRUT 7075-T6 AL TUBE
MONOPROPELLANT ENGINE
SECTION D - D
TANK SUPPORT CONE 7075-T6 AL SKIN, FRAMES & STIFFENERS
AFT FRAME 7075-T6 AL FORMED SECTION
A



DEPLOYED POSITION OF SOLAR PANEL
OPEN POSITION OF POWER PANEL (TYPICAL FOR COMMUNICATION PANEL & EXPERIMENT EQUIP PANEL
CENTAUR/SPACECRAFT ADAPTER CONE SEMI-MONOCOQUE CONSTRUCTION SKIN, FRAMES & INTERMEDIATE STRINGERS 7075-T6 AL ; TAPERED MAIN STRINGERS 7075-T73 AL FORGINGS
SOLAR PANEL STIFFENING MEMBERS
ON POWER PANEL
FOR SOLAR PANELS
SOLAR PANEL 1" THK HONEYCOMB SANDWICH 1.6 #/FT³ CORE DENSITY .010 THK FACES 7075-T6 AL
ANTENNA LATCH (PULL PIN FOR RELEASE)
SOLAR PANEL (STOWED POSITION)
STOWED SOLAR PANEL
B
EQUIPMENT MOUNTING PANEL 1" THK TRUSS GRID SANDWICH 3.1 #/FT³ CORE DENSITY .025 THK FACES 7075-T6 AL
EQUIPMENT SUPPORT RAILS .100 THK 7075-T6 AL EXTRUDED
45°
MOUNTING FACE FOR GIMBAL MECH.
3/8" DIA BOLT & SEPARATION NUT 4 PLACES 90° APART
HANDLING LUG ℄
DEPLOYED POSITION OF HIGH GAIN ANTENNA
TANK (2)
VIEW B-B
NOTE: THIS VIEW DOES NOT SHOW ANTENNA OR SOLAR PANELS
C-C
CORNER MEMBERS
0 10 20 30
SCALE IN INCHES
TRW SPACE TECHNOLOGY LABORATORIES
THOMPSON RAMO WOOLDRIDGE INC.
ONE SPACE PARK · REDONDO BEACH, CALIFORNIA
MARS VOYAGER - PHASE IA, 1969 TEST, A3 STRUCTURAL CONFIGURATION
20 JULY 1965
PD85-78


Figure 5-4.

### 7.3 Midcourse Propulsion Module

The midcourse propulsion module consists of a horizontal panel at spacecraft station 62 plus tank and engine support structure. The panel is a square 58 inches on a side. It attaches to the lower bus frame with an interchangeable bolt pattern. Sandwich construction is used for the panel. It has a 1-inch-thick aluminum truss grid core with 0.025-inch 7075-T6 aluminum skins. In addition to serving as a structural member it also provides meteoroid protection to the aft end of the spacecraft. There is a support structure for the pressurant/propellant tank and two attitude control tanks which mount on the panel. The propellant tank is located on the centerline and rests on a transverse cone-shaped flange on the tank support cone. Two straps at 90 degrees to each other go across the top of the tank and attach to the base of the tank support structure with tensioning devices. The midcourse engine mounts to a smaller cone located on the centerline of the panel. The support for the two attitude control tanks is similar to that for the propellant tank and mount on the tank support cone.

### 7.4 Solar Panel Structure

The solar array consists of three identical deployable panels and the supporting structure. The panels are in the form of a trapezoid. They are constructed of 1-inch-thick honeycomb with a standard aluminum honeycomb core and a 0.010-inch 7075-T6 aluminum skins. Support beams run down the edges of the panel. Hinges are located at the aft end of each support beam. The hinge points on the bus attach to the bus frame at spacecraft station 56. The panel deployment system is described in V. 12.

### 7.5 High Gain Antenna Support

The high gain antenna support structure is in the form of a support truss with two tubes running to adjacent corners of the bus at spacecraft station 62.

### 7.6 Voyager/Centaur Interstage Structure

The interstage structure consists of a 57-inch-diameter cylinder between spacecraft stations 0 and 12, and a truncated cone tapering from the 57-inch diameter at spacecraft station 12 to an 87-inch diameter at the spacecraft separation plane (spacecraft station 53). Construction consists of 7075-T6 aluminum skin and stringers with frames at spacecraft stations 0, 12, 32 and 53. A cutout is provided at the upper end to clear the antenna support arm. Four of the stringers have integral end fittings which attach to the four corner fittings of the spacecraft. Attachment to the Centaur is a bolted field joint.

### 7.7 Meteoroid Protection

In regard to meteoroid protection, the surface area of the 1969 spacecraft is less than that for the 1971 spacecraft, but the transit time is typically longer. As shown in Appendix C of Volume 5, the probability of no punctures for the 1969 test spacecraft is about the same as that of the 1971 spacecraft.

## 8. STRUCTURAL DESIGN CRITERIA

The basic requirements and criteria for the structural design and testing of the 1969 Voyager spacecraft will be the same as those for the 1971 Voyager spacecraft as specified in Volume 21, VS-4-521, except that the load conditions and vibration environments for the Atlas-Centaur vehicle launch phase will be used.

The launch phase is defined to include liftoff, flight through the atmosphere, Atlas burnout (BECO), Centaur first burn and Centaur second burn.

The dynamic structural interaction of the launch vehicle and the spacecraft will be considered in the launch phase analyses of the composite spacecraft design loads. The acceleration values during the launch phase will be based upon rational analyses; and the loads upgraded by an iterative approach throughout the design phase.

Design flight limit load factors induced by the Atlas-Centaur vehicle are defined below for the flight phases expected to be most critical for the structure subsystem.

| | | |
|---|---|---|
| Max. q | +2.2 | 1.72 |
| Max. acceleration (BECO) | +5.9 | .45 |
| Centaur first burn | +1.0 | 1.30 |
| Centaur second burn | +2.2 | 1.30 |
| Centaur max. acceleration | +4.0 | .80 |

Notes: (1) The longitudinal and lateral load factors will be applied simultaneously, and are assumed to act at the center of mass of mass of the spacecraft.

(2) The longitudinal direction refers to the thrust axis of the boost vehicle. A positive factor indicates the load is acting aft.

(3) The lateral direction is any direction perpendicular to the thrust axis.

An estimate of the longitudinal vibration environment covering all events from liftoff to spacecraft separation is as follows:

Sinusoidal Part of Vibratory Input

| | |
|---|---|
| 5 to 11 cps | 0.25 inch single amplitude (zero to peak) |
| 11 to 2000 cps | 3 g vibratory acceleration (zero to peak) |

Random Part of Vibratory Input (Gaussian distribution)

| | |
|---|---|
| 50 to 300 cps | 0.001 $g^2$/cps at 50 cps increasing linearly to 0.02 $g^2$/cps at 300 cps |
| 300 to 100 cps | 0.02 $g^2$/cps (constant) |
| 1000 to 2000 cps | 0.02 $g^2$/cps at 100 cps decreasing at 12 db/octave to 2000 cps |

## 9. PROPULSION

The weight and volume constraints for the 1969 mission preclude the incorporation of the 1971 retropropulsion solid motor. However, the 1971 midcourse propulsion subsystem and the evasive maneuver propulsion can be validated by the 1969 flight test. The electrical interface with the retropropulsion motor can also be carried and verified as a test installation.

### 9.1 Midcourse Propulsion

The 1969 midcourse propulsion operates in the same manner and has the same configuration as that for 1971, except for the removal of one of the two pressurant propellant tanks and the associated changes in plumbing. The modular configuration is maintained and the flight-ready system is incorporated into the spacecraft as part of the midcourse propulsion module described in V.7. The single propellant tank is located on the centerline directly forward of the engine as shown in Figure 4-2. Midcourse propulsion components are listed in IV.6 along with weight, power, and temperature parameters. A functional description is given in VS-4-610.

For a nominal 1969 spacecraft weight of 1500 pounds, a single fully loaded propellant tank is sufficient to impart a velocity of 195 meters/sec, as compared with 75 meters/sec for the two tanks in the 1971 configuration. Midcourse propellant for the nominal 1969 configuration has been off-loaded to correspond to 75 meters/sec, but this loading can be increased if an additional propulsive capability is desired.

The achievable accuracy for a velocity increment is inversely proportional to vehicle mass. Hence the 1969 midcourse propulsion system is capable of delivering a minimum midcourse velocity increment of 0.5 meter/sec with a 3-sigma nonproportional error of 0.05 meter/sec, with these quantities approximately five times those for the 1971 system. The over-all accuracy realized in the system application, however, depends on the timing accuracy achievable for start and stop of the engine firing.

### 9.2 Evasive Maneuver Propulsion

A blow-down cold gas system is utilized in the 1971 mission to provide a propulsive capability for an evasive maneuver. This is carried out after capsule vehicle separation to avoid interference between the separated vehicles. Essentially the same unit for the 1971 system can be incorporated into the 1969 configuration for test purposes if desired. Installation of the unit is shown in Figure 4-2 and a functional description is given in VS-4-612.

The operation of the 1971 equipment can be validated during the 1969 mission. The total evasive maneuver, however, involves stability and control under the lateral translation and, hence, depends on spacecraft dynamic characteristics and alignments. Although these aspects will be different for the two systems, it is still possible to apply the 1969 test results to the 1971 mission through use of analytical modeling.

## 9.3 Solid Propellant Motor Simulation

Whereas the 1969 Voyager flight test vehicle provides an excellent opportunity to evaluate the midcourse propulsion system, a similar check on the retro motor does not appear to be feasible. Some small additional increment of confidence in certain solid motor components might be achieved by the firing of a scaled-down motor on the 1969 flight test vehicle; however, this increment does not appear to justify the development of a special motor.

An alternate approach would be to use a fully developed motor such as the Aerojet Cygmus-15, adapted with a new LITVC system. This approach could be achieved within the available development time and the effort spent on developing the LITVC system would be applicable to the retro motor for the 1971 flight. However, the major potential problem areas involved in the solid motor design nozzle life, propellant performance with low burning rate propellants, and case strength degradation resulting from prolonged exposure to vacuum are not directly applicable and can probably be better simulated and evaluated in ground tests. Retro motor-vehicle interactions such as exhaust plume heating and contamination of the space vehicle are difficult to simulate on ground tests. However, evaluation of these interactions resulting from the firing of a solid motor on the 1969 test vehicle would be even more difficult.

The conclusion is that simulation of a solid propellant retro motor for the 1969 mission is not recommended.

## 9.4 Solid Motor Component Tests

It appears feasible and desirable to simulate the 1971 solid motor-vehicle electrical interface in the 1969 flight test. This amounts to an igniter circuit and an applicable test installation of the 1971 liquid injection thrust vector control (LITVC) components.

The proposed LITVC installation would include the solid propellant gas generator, the Freon storage vessel (off-loaded if weight becomes critical), and one fluid injector valve and actuator. The three other injector units would be simulated with dummy black boxes. Commands to the LITVC actuator could be generated by the CS and C and the Freon could be vented overboard with either a nonpropulsive device or with a device which perturbs the vehicle attitude. The second approach would provide an interaction with the reaction jet system and provide data on the stabilization and control subsystem performance. The hardware weight required to test both the igniter circuit and the LITVC components would be 40 to 50 pounds.

# 10. PYROTECHNICS

To a maximum extent the pyrotechnic devices of the 1971 spacecraft will be flight tested in the 1969 mission. This will include not only the electrical firing circuits and ordnance items but the launch preparation and pyrotechnic checkout procedures. A functional description of the 1971 pyrotechnic items is given in VS-4-530.

Subject to differences in boost powered flight environment, physical access and spacecraft configuration, the 1969 flight test will validate the design of the 1971 pyrotechnic elements.

## 10.1 Spacecraft-Launch Vehicle Electrical Umbilical Disconnect

The electrical connection between the spacecraft and the launch vehicle required for ground checkout and support will be separated before launch by the same design of electro explosive disconnect device used in the 1971 mission.

## 10.2 Spacecraft-Launch Vehicle Separation

The explosive separation nuts used in the 1969 flight test spacecraft-launch vehicle separation system are the same design as those used in the 1971 mission. In the 1969 system, four separation nuts are used instead of three separation nuts. The separation signal is routed to the separation devices from the Centaur in the same manner as for the 1971 system.

## 10.3 High Gain Antenna Dish Deployment

The pin puller and two EED which release the high gain antenna dish for deployment have the same design as those used for release of the high gain and medium gain antennas and planet-oriented package on the 1971 spacecraft.

## 10.4 Midcourse Correction Motor Control

The explosively actuated valves and their EED used to control the midcourse correction motor on the 1969 flight test spacecraft are identical to those valves and EED to be flown on the 1971 mission.

## 10.5 Solar Panel Release

The folded solar panels used on the 1969 flight spacecraft will be dereefed for deployment by the action of either of two parallel pyrotechnically actuated guillotines. This subsystem has no counterpart in the 1971 spacecraft design.

## 10.6 Evasive Maneuver Propulsion Actuation

The evasive maneuver is performed by the 1969 system just as in the 1971 mission. A 1971 type normally-closed explosively actuated valve is opened by the firing of one or both of two pressure output type EED.

## 10.7 Jettison of Capsule Adapter and Canister

A design verification test of the capsule adapter and canister separation components will be made on the 1969 spacecraft. The electro-explosive devices, confined detonating fuse leads, and explosive separation nuts of the design to be used on the 1971 spacecraft will be included as a test installation. This subsystem will be installed and operated in the 1969 system in essentially the same manner as for the 1971 mission.

## 10.8 Retropropulsion Pyrotechnic Elements

It is feasible to incorporate a test installation of the retropropulsion igniter and the LITVC solid propellant gas generator with its igniter into the 1969 spacecraft. This is recommended as discussed in V.9 and will validate the corresponding pyrotechnic elements for the 1971 system.

# 11. LAUNCH VEHICLE-SPACECRAFT SEPARATION

The 1969 launch vehicle-spacecraft separation mechanism is essentially the same and operates in the same way as that for the 1971 system. It is worth noting that the Centaur is utilized for 1969 and 1971 and can be caused to separate in the same manner for both missions. The separation installation at spacecraft station 53 is shown in Figure 5-4. A functional description of the 1971 system is given in VS-4-570.

The 1969 configuration leads to four separation devices rather than three as on the 1971 spacecraft. In the 1971 system the separation bolts take tension loads only, with shear loads across the separation joint taken by three shear pins. Similarly, for the 1969 system, except that instead of three separate shear pins, there are four such pins and they are mounted concentrically around each separation bolt as shown in Figure 5-5. The pyrotechnic device is the same for 1969 and 1971 and is described in V.10.

The basic similarity between the separation systems for 1969 and 1971 as discussed above leads to the conclusion that the 1969 flight test can validate the 1971 separation system.

# 12. SOLAR PANEL DEPLOYMENT

The solar panel deployment subsystem is used to release and move the solar panels at a controlled rate through an arc of approximately 100 degrees from the latched position to the deployed position. Since the selected 1971 spacecraft configuration has a fixed solar array, there is no relationship between the subsystem described here and the 1971 spacecraft.

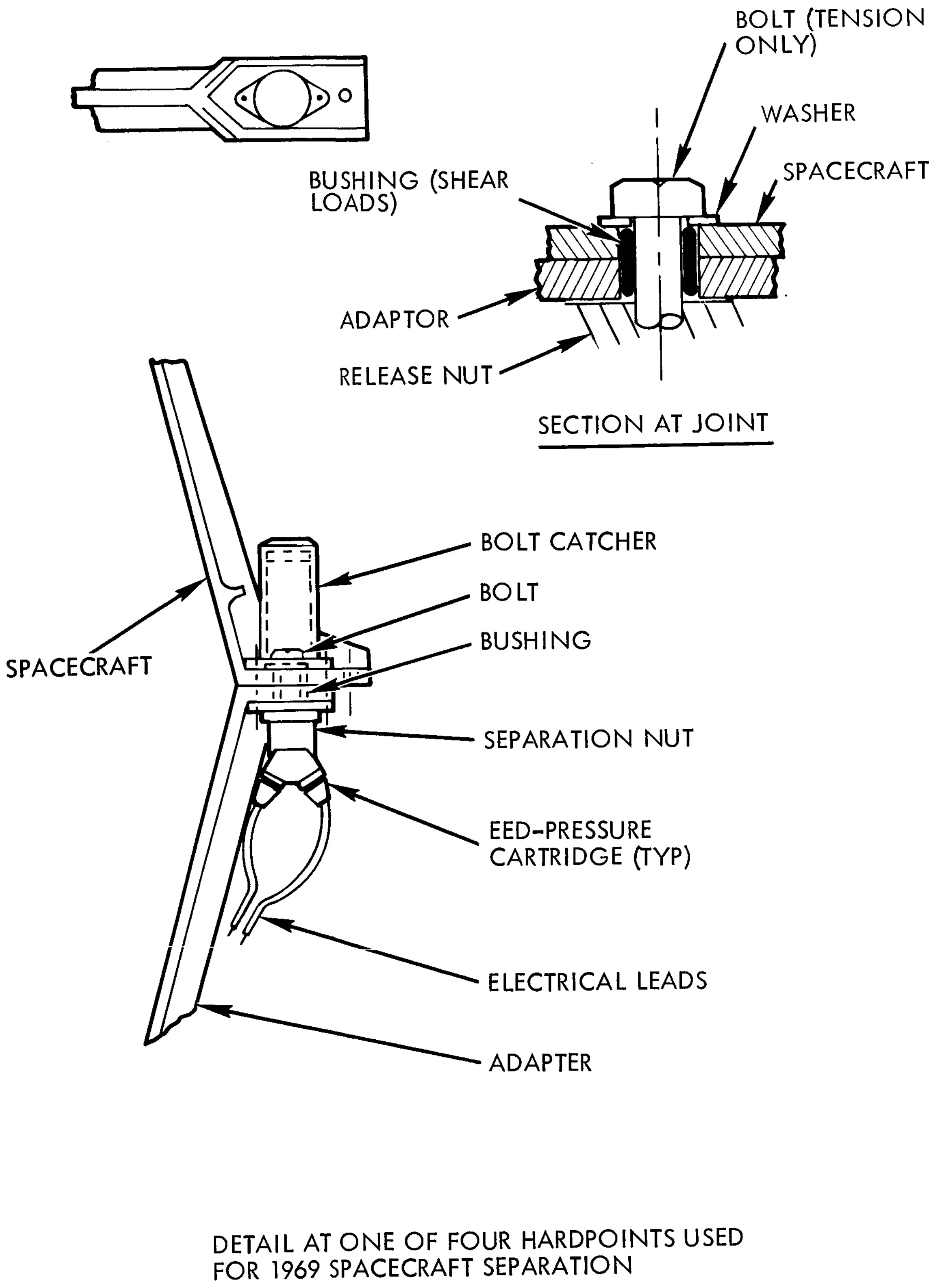



Figure 5-5. 1969 Launch Vehicle-Spacecraft Separation Device

The solar panel deployment subsystem consists of the following four major elements (see Figure 5-6):

- o Three unlatch mechanisms (one for each solar panel), two nylon cords, that hold all three latches in the restrained position, and two pyrotechnic bolt cutters.
- o Two hinge line springs on each panel to provide the opening torque for the solar panel.
- o A hydraulic retarder to provide positive control of the angular rate of each solar panel.
- o A mechanical lock to hold each panel in the extended position.

The design of the latch system will be such that the nylon cord, once cut, will remain in the initial position except for enough tension relief to allow latch release. Release latches and cable cutters will be mounted to structure and will be independent of the solar panels. The actuation springs and dampers will be integrated with the solar panel hinges.

The two deployment springs will be capable of exerting the following nominal total static torque about each solar panel hinge axis.

| | |
|---|---|
| Stowed position | 50 in-lb |
| Deployed position | 25 in-lb |

The retarder will be capable of controlling the angular rate of solar-panel deployment so that the time for full extension is 30 to 60 seconds.

The solar panel deployment subsystem will be designed for a minimum life of 75 cycles without degradation of performance.

The total electrical power required for the two squibs (two bridgewires per squib) will be 20 amperes, 28 volts, for a 0.050 second duration.

The total weight of the actuation subsystem including the release, actuation, retarding, and locking mechanism will not exceed 4 pounds.

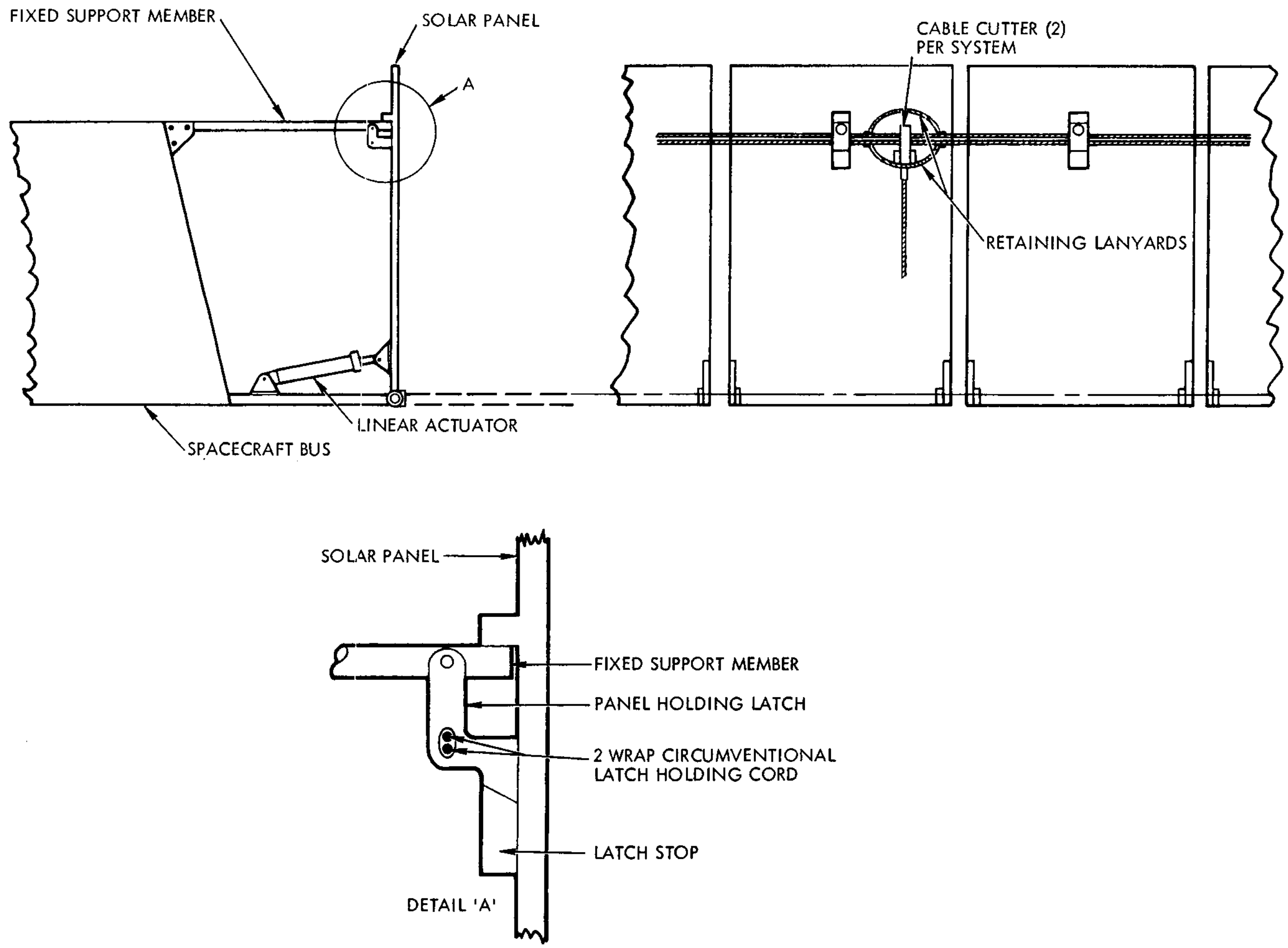



Figure 5-6. Solar Panel Deployment

# VI. IMPLEMENTATION PLAN

The implementation plan for the 1969 Voyager test spacecraft is considered an intermediate development phase toward the ultimate goal of attaining a reliable 1971 orbital mission design. The same tasks of engineering development, manufacturing, assembly and checkout, spacecraft testing, and launch operations are applicable to both. Thus, the complete development of this flight test spacecraft is discussed in Volume 3, Section V.

This section summarizes the development primarily by highlighting the significant difference and conversely the common elements of design of the two spacecraft.

## 1. STRUCTURAL SUBSYSTEM

The structure for the 1969 test flight spacecraft differs from that of the 1971 spacecraft because of the constraints of weight and usable volume imposed by the Atlas-Centaur launch vehicle. The basic structural arrangement consists of a structural frame, similar in principle to that for the 1971 spacecraft, supporting four (instead of six) equipment panels. The adapter section to the Centaur differs from the 1971 version; it contains the separation system and encloses the double gimballed high-gain antenna. The three solar array panels are separately mounted and deployed. The monopropellant midcourse engine, tankage, and the stabilization and control pneumatics are contained in a separable module in much the same manner as for the 1971 design. The electronic equipment mounting technique is identical to that for the 1971 design, and, in fact, three of the mounting panels and their associated equipment can be identical between 1971 and 1969. The fourth panel of the 1969 system would mount whatever science equipment is desired and would probably, but not necessarily, differ from its 1971 counterpart.

Thus, the 1969 structural subsystem requires a separate development effort although the design approach is retained using loads criteria based on the Atlas-Centaur rather than the Saturn-Centaur.

The implementation plans for both designs are discussed in Volume 3, Section V, reflecting a 27-month program from the start of Phase 1B. This permits an additional 12 months for spacecraft assembly, test, and launch operations to accommodate the test flight. Phase 1B will be devoted to completing initial layouts; design freeze is accomplished within 1 month after Phase II start. Structural model tests require completion approximately 13 months after Phase 1B go-ahead.

The structural design is easier than for the 1971 spacecraft due to the deletion of the retropropropulsion and the capsule separation system. In addition, the 3 foot circular parabolic medium gain antenna and experiment interfaces are not required. Some design complication is introduced due to the solar panels which are folded and deployed, requiring additional hinge and latch mechanisms.

The key structural advantage for the 1969 flight results in flight environment experience for the electronic equipment panels, high gain antenna, and drive and monopropellant tank and tank mounting.

The design development task for the structural subsystem as discussed in Volume 3, applies to the 1969 test flight.

2. THERMAL CONTROL

The total spacecraft thermal design requires a supplemental effort for the 1969 flight. Much of the information obtained during the early development in terms of thermal control subassembly design and performance will be applicable to the 1971 mission design. Identical thermal control louver assemblies are planned for both flights; the solar panel thermal control is the same for the module design, the total area and geometry differ; thermal control requirements resulting from retro-propulsion needs are eliminated; and the balance of the spacecraft (excluding panel mounted electronics) requires separate analysis and thermal control design.

The activities planned for design and development of the 1969 Voyager thermal control subsystem are contained in Volume 3, Section V, Paragraph 4.2, including the implementation schedule. The thermal design for 1969 is considered an intermediate development for the eventual 1971 mission.

## 3. PROPULSION SUBSYSTEM

The midcourse propulsion subsystem (MPS) utilizes nearly the identical design for both 1969 and 1971 flights. The number of tanks is reduced from two to one, but the tanks are identical. The plumbing is modified, but the valving and engine are identical. The single tank is off loaded so as to provide a 75 meter/sec capability. The approach to the MPS develops the subsystem early in the program permitting maximum reliability testing to proceed before either flight. The complete MPS development plan for 1969/1971 is discussed in Volume 3, Section V, Paragraph 4. 3. 1 as the 1969 program is considered completely common to both flights.

The retropropulsion system discussed in Volume 3 is not required for the 1969 flyby mission. Should the test flight objectives change to include orbital operations for 1969, considerations will then be given to retropropulsion; however, with the weight constraints associated with an Atlas-Centaur boost vehicle this does not seem probable.

## 4. STABILIZATION AND CONTROL

The 1969 flight will utilize generally the same stabilization and control subsystem as the 1971. The inertial and optical guidance sensors, midcourse propulsion thrust control and reaction control equipment are the same. Slightly different requirements are imposed upon the equipment due to geometry and mas properties changes between the two vehicles. The reaction control differs slightly because of solar panel locations (deployable three-panel arrangement) and the high-low thrust roll orientation nozzles will be flown only for test purposes. Thus, the development of the 1969 flight equipment is extended toward the ultimate requirements of the 1971 mission. The implementation plan for this subsystem is discussed in Volume 3, Section V, Paragraph 4. 4 including the analysis, design, test, and schedule proposed.

## 5. CENTRAL SEQUENCING AND COMMAND (CS&C)

The CS&C subsystem is identical for both 1969 and 1971 flights except that the command associated with the experiments and sensors may be blind ended back to telemetry for the experiments not on board. As the 1969 test flight minimizes the experiments to be flown, additional capacity is available for additional engineering instrumentation. The same equipment will be flown for both flights and thus the CS&C development is essentially completed during the 1969 subsystem development. Detailed application changes may require slight modifications and subsequent design releases for 1971. The complete CS&C subsystem implementation plan is contained in Volume 3, Section V, Paragraph 4.5.

## 6. COMMUNICATIONS AND DATA HANDLING SUBSYSTEM

A maximum use of the 1971 communications and data handling equipment will be flight tested in 1969 and developed early to accomodate this schedule. This will include the reliability redundance aspects. The equipment which will not require early development includes the 3-foot circular parabolic medium gain single gimbal antenna and the VHF capsule antenna. The payload envelope will not permit the incorporation of the 3-foot dish and therefore a second identical S-band cup turnstile low-gain antenna will be utilized for antenna coverage in its place. The capsule is not considered part of the 1969 configuration, although a small capsule could be carried, and therefore its antenna is not required. Other significant changes are associated with data handling for the 1971 mission experiments; however, redundant tape recorders are planned for the 1969 mission. The balance of the equipment planned for 1971 will be tested during the 1969 flight. The data rate can be simulated by incorporating PN generator to simulate the science data and check out the data handling system. The development plan for the 1969 test is shown in Volume 3, Section V, Paragraph 4.6.

## 7. POWER SUBSYSTEM

The 1969 test flight will employ the same power subsystem equipment as used on the 1971 flight except the solar arrays are necessarily configured differently. However, the solar cell module design is completely retained and should result in valuable test data directly applicable to the 1971 mission. The solar arrays, solar array hinges and structure will be new for 1969. Otherwise identical equipment will be developed early for the 1969 test and results in direct application to 1971 mission requirements. The development plan for the 1969 power subsystem is included in Volume 3, Section V, Paragraph 4.7.

## 8. ELECTRICAL DISTRIBUTION SUBSYSTEM

The difference in general arrangement and deletion of certain equipment for the 1969 spacecraft require a change in the design of the electrical distribution subsystem from that for the 1971 spacecraft. The major components are utilized for both designs and the design and development are identical for both spacecraft. Differences arise from the deletion of the planet-oriented package experiments, body mounted experiments, retropropulsion engine and medium gain antenna. The requirement for spacecraft engineering instrumentation will also impose an additional requirement.

The development plan for the 1969 effort is included in Volume 3, Section V, Paragraph 4.9 as an integrated effort which evolves into the design for the 1971 spacecraft.

# VII. OSE OBJECTIVES AND CRITERIA

Since it is anticipated that the OSE for the 1969 Voyager test program will be identical or so similar that only minor modifications will be required to the 1971 Voyager OSE, the OSE objectives and criteria are the same for the two programs except for minor change in the MOSE. The OSE objectives and criteria contained in Section I, Volume 6 of this report are directly applicable to the 1969 Voyager OSE. This data will be documented into specification format following more detailed discussions with JPL. The following paragraphs are highlights of the OSE objectives and criteria data included in volume 6.

The OSE provides for the highest practical probability that at the time of launch the Voyager mission will succeed in all of its objectives. In providing this objective the OSE must be capable of verifying spacecraft design and detecting spacecraft faults, does not prevent launch of a proper performing spacecraft, and provides the most expeditious techniques for remove, repair and/or replace, retest, and remate on stand.

## 1. ELECTRICAL OSE (EOSE)

The EOSE consists of checkout equipment which will demonstrate the proper design of the various flight units, flight subsystems, and the integrated spacecraft. It supports all facets of the Voyager test operations as well as the types of tests which will detect design deficiencies in the flight hardware early in the program. The accumulation of effective test data and the establishment of proper test parameters is an important criteria for EOSE design. Written test procedures will be used at all test locations including factory, field, and launch site. No malfunction will remain unexplained and in the event troubleshooting is required all steps will be documented. Test times for all phases of the test program will be performed in the most expeditious but most effective manner.

## 2. MECHANICAL OSE (MOSE)

All AHSE is fabricated of nonmagnetic materials and is designed to provide maximum shock attenuation and vibration damping. All transportation AHSE is designed for acceptance aboard feasible transportation media. All MOSE is designed to withstand proof load testing and MOSE which performs tasks of handling, transferring, and shipping the spacecraft provides design features which adequately satisfy environmental constraints imposed by the spacecraft system.

# VIII. OSE DESIGN CHARACTERISTICS AND RESTRAINTS

## 1. GENERAL

Because of the anticipated similarity between the 1969 test spacecraft and the 1971 spacecraft, the design characteristics and restraints for the OSE required for each of these programs are basically the same. The 1971 Voyager OSE design characteristics and restraints contained in Section II of Volume 6 are directly applicable to the 1969 Voyager OSE with the exception of any reference to the capsule and its related OSE.

The following paragraphs are highlights of the design requirements and constraints discussed in Volume 6.

This section defines the design criteria and restraints which will be applied to the operational support equipment necessary to support the Voyager 1969 mission test and evaluation program. Included are those design requirements which will be applied to the design, fabrication and checkout of the system test complex (STC), launch complex equipment (LCE) and mission dependent equipment (MDE).

## 2. ELECTRICAL OSE REQUIREMENTS

The basic capabilities of the EOSE are that the inputs and outputs of all flight equipment under test will be monitored for quantitative evaluation of performance. In order to accomplish this, sufficient monitor points will be provided, sufficient isolation from the equipment under test will exist, and power will be provided to the flight equipment.

Unit test sets are capable of providing the functions of checking out the units of a subsystem, an assembled subsystem, and evaluation of the equipment on a spacecraft panel.

The system test set (STS) is used primarily in the evaluation of proper operation of the spacecraft. This evaluation is accomplished by performing tests on the various spacecraft subsystems in all operating modes.

The requirements of the launch control equipment (LCE) are identical to the STS except for geographical location. A ground power console and ground power rack are located at the blockhouse and at the explosive safe facility.

The mission dependent equipment (MDE) is provided to all Voyager DSIF stations. It is designed to provide the general functions of command generation and detection, telemetry detection, computer buffering, spacecraft status display, testing the MDE, and spacecraft and DSIF simulation.

## 3. MECHANICAL OSE REQUIREMENTS

Provision is made in the design of all mechanical operational support equipment to insure that loads encountered during conditions of assembly, handling and shipping do not control the design of the spacecraft or any component, to the extent that additional flight weight is required.

The MOSE is designed to withstand the application of limit loads without permanent deformation or excessive deflection. Excessive deflections are those which result in unsatisfactory mechanical performance or induce loads in the spacecraft or components that exceed the design loads.

The MOSE is designed to withstand design ultimate loads without failure. Failure is defined as inability to sustain ultimate load. Material strengths and other physical properties are selected from reliable test results of recognized laboratories, reports from government agencies or manufacturer's guaranteed data.

# IX. OSE SYSTEM FUNCTION DESCRIPTIONS

## 1. SYSTEM ELECTRICAL OSE

This section contains data on system level EOSE, that is, EOSE used to support system level testing of the Voyager test spacecraft and its associated supporting equipment. Except for minor differences in panel detail, this equipment is identical for both 1969 and 1971 Voyager missions.

### 1.1 System Test Set (STS)

The STS is used for integrated system testing of the spacecraft during integration assembly and testing, testing in the environmental area, and testing at ETR in the spacecraft assembly facility (see Figure 9-1).

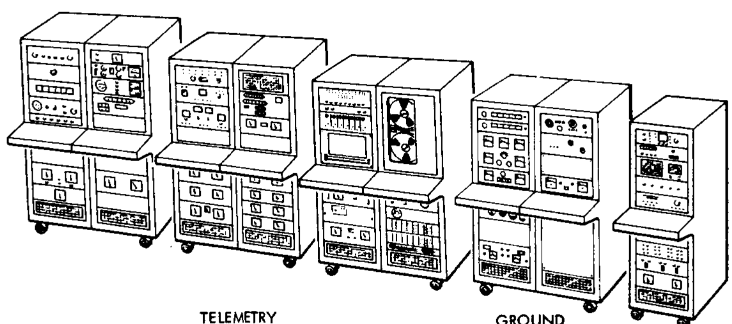



Figure 9-1. System Test Set

Tests at the propulsion test site (Capistrano), at the magnetic facility (Malibu), and in the environmental areas of TRW Systems will be supported by an STS in the Voyager assembly facility of TRW Systems, but with transfer of the RF console and the ground power consoles to the vicinity of the spacecraft. Similarly, the STS is used in the spacecraft assembly facility for tests in that facility, and to support other tests in the ETR such as tests in the explosive safe facility and on the launch pad, where again the RF consoles and the ground power consoles are grouped near the spacecraft under test while the system test set remains in the spacecraft assembly facility. In all cases, when the RF consoles and ground power consoles are used at remote locations, data flow between these units and the remainder of the system test sets is by direct video cable or by wideband digital relay. See OSE/US-3-110 in Appendix G of Volume 6.

## 1.2 Automatic Data Handling System (ADHS)

The ADMS is located with the STS to support system tests conducted by the STS during spacecraft tests in the TRW Systems Integration Assembly and Test area and at the ETR in the SAF, the ESF, and on the launch pad. The ADHS consists of a test director's console, an SDS-930 computer, manual input devices for transmitting data from the STS or associated equipment in the ESF, the blockhouse, or the launch pad, and computer peripheral equipment such as tape stations, line printers, character printers, paper tape punches and readers, etc. See OSE/VS-3-120 in Appendix G of Volume 6 and Figure 9-2.

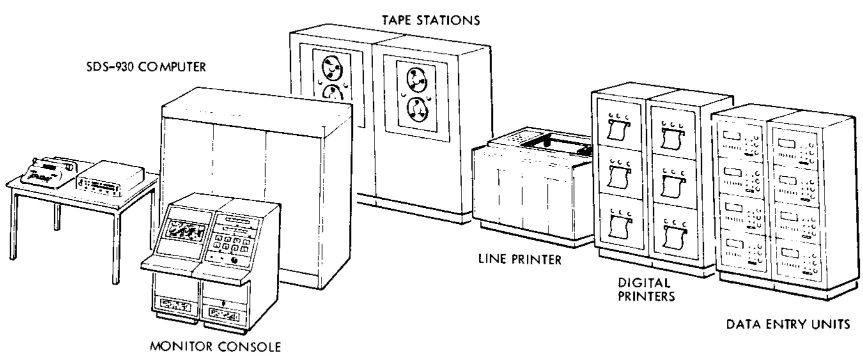



Figure 9-2. Automatic Data Handling System

## 1.3 Launch Complex Equipment (LCE)

The LCE is used at ETR to support testing at the SAF, ESF, launch pad, and blockhouse. See OSE/VS-2-120 in Appendix G of Volume 6.

### 1.3.1 Spacecraft Assembly Facility (SAF)

The system test set (STS) is used to conduct spacecraft tests in the SAF and to support tests at remote locations in the ETR, such as the explosive safe facility and on the launch pad. Additionally, an automatic data handling system (ADHS) is used to support real time evaluation and recording of pertinent checkout data in conjunction with the STS (see Figure 9-3).

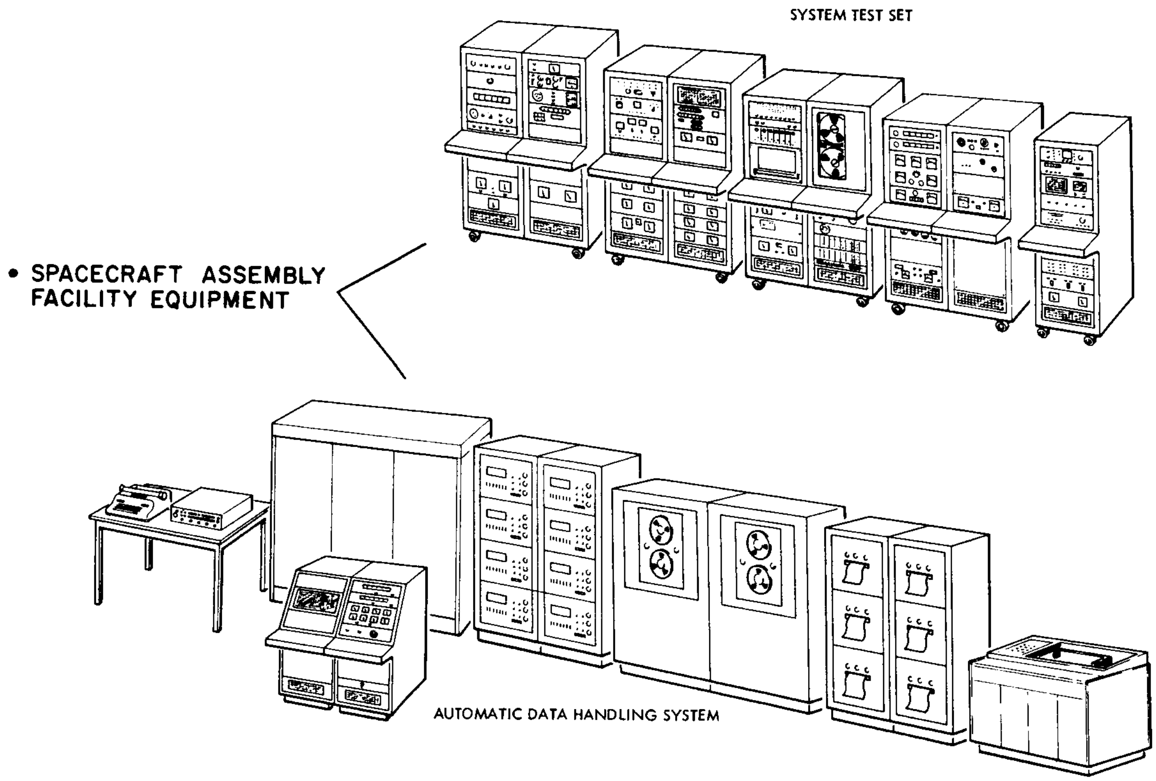



Figure 9-3. Spacecraft Assembly Facility Equipment

### 1.3.2 Explosive Safe Facility (ESF)

Installation of the capsule and ordnance are checked out in the ESF using the RF consoles and ground power consoles of the STS which are located in the SAF. To provide local display capability, a duplicate blockhouse monitor console is included in the ESF complement (see Figure 9-4).

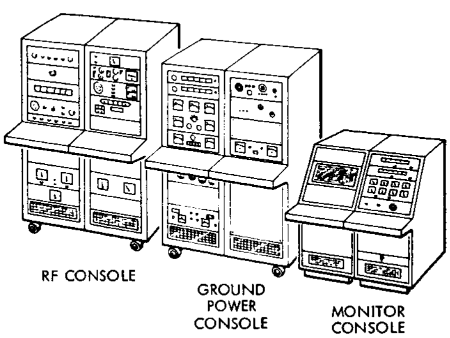



Figure 9-4. Explosive Safe Facility Equipment

1.3.3 Launch Pad Equipment

Launch pad equipment consists of the ground power consoles and the in-flight jumper control equipment used in connection with the STS in the SAF and with the monitor console in the blockhouse (see Figure 9-5).

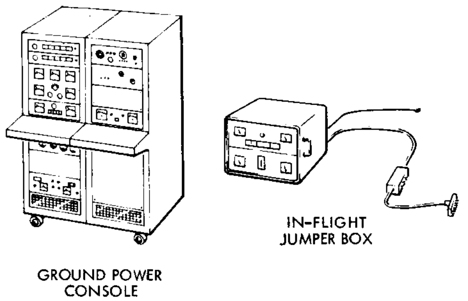



Figure 9-5. Launch Pad Equipment

1.3.4 Blockhouse Monitor Console

The blockhouse monitor console provides control and display of the status of the power subsystem and display of spacecraft and telemetry status. Inputs for driving the blockhouse monitor console are derived from the ADHS in the SAF and from hardlines from the launch pad (see Figure 9-6).


Figure 9-6. Blockhouse Monitor Console

1.4 Mission Dependent Equipment (MDE)

MDE at the DSIF includes in-line equipment such as telemetry detectors, computer buffering, and command generation, and supporting test equipment such as transponders, data format generators, error rate testers, command detectors, etc. See OSE/VS-3-130 in Appendix G of Volume 6 and Figure 9-7.

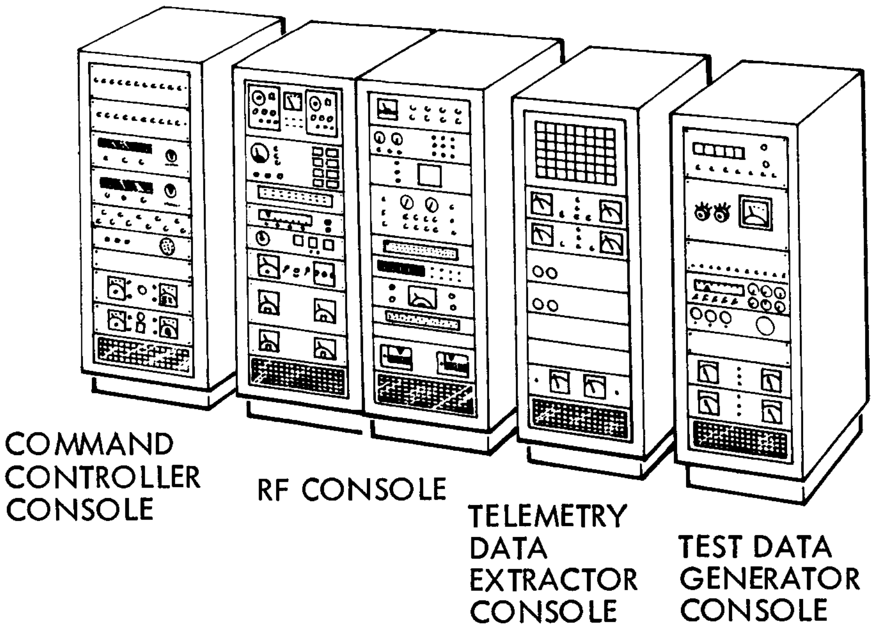



Figure 9-7. Mission Dependent Equipment

2. SYSTEM MECHANICAL OSE

2.1 General

This section defines the assembly, handling, and shipping equipment (AHSE) required for the assembly, checkout, handling, and transport of the 1969 test spacecraft. Because of equipment similarities and the desire to use common mechanical support equipment for both 1969 and 1971 spacecraft systems, the referenced equipment is related to, and is to be interpreted with OSE/VS-3-140 and all its related equipment documents for the 1971 system. (Reference Voyager Phase IA Study Report, Volume 6, Section III.)

To distinguish differences between the 1969 and 1971 Voyager mission OSE when they do exist, the equipment described in this section is identified as the TVS-3-140 series.

All applicable documents, general requirements, and specific equipment requirements are the same for both the 1969 and the 1971 mechanical operational support equipment (MOSE) except as noted herein. The equipment requirements for the 1969 and 1971 systems are compared in the 1969/1971 MOSE comparison matrix shown in Table 9-1).

Following is a series of notes indicating differences between this equipment (for 1969) and the 1971 Voyager mission. During Phase IB full specifications will be generated.

## 2.2 Equipment Description

### 2.2.1 Transporter Test Spacecraft (TVS-3-140-1)

The 1971 Voyager spacecraft transporter may be used to transport the 1969 test vehicle. However, the 1969 test vehicle is transported fully assembled and because of the test vehicle dimensions, it is shipped horizontally. A cradle adapter (TVS-3-140-19) is required that mates with the spacecraft's upper structural plane and the Centaur's interface plane.

The cradle mounts to the 1971 spacecraft mounting points. Additional support structure is required on the transporter to support and brace the solar array panels in their folded attitude. Because of the horizontal shipping sttitude, additional shock mitigation is required on the transporter to reduce loads imposed on the flight spacecraft to acceptable limits.

### 2.2.2 Assembly, Handling and Tilt Fixture (TVS-3-140-2)

The 1969 test spacecraft mounts on the 1971 assembly, handling, and tilt fixture by providing an adapter section that interfaces with the tilt fixture circular mounting ring and the 1969 test spacecraft mounting interface plane. This adapter allows the fixture to mount a different size and shape spacecraft than it is primarily designed for.

### 2.2.3 Transport Recorder (TVS-3-140-3)

The same transport recorder is used for either spacecraft. The sensing elements are attached at appropriate locations on the 1969 test

Table 9-1. 1969/1971 MOSE Comparison Matrix

| 1971 Equipment | | 1969 Application of 1971 Equipment | 1969 Functional Requirements | 1969 Adaptation | | | All New Equipment Required |
|---|---|---|---|---|---|---|---|
| Item No. | Nomenclature | | | Use As Is | Add Mod Kit | Modify Equipment | |
| 3-140-1 | Transporter, Flight Spacecraft | Yes | Same | Yes | Yes | - | - |
| 3-140-2 | Assembly, Handling, and Tilt Fixture | Yes | Same | Yes | Yes | - | - |
| 3-140-3 | Transport Recorder | Yes | Same | Yes | - | - | - |
| 3-140-4 | Fixture, Weight, CG, and MOI | Yes | Same | - | Yes | Yes | - |
| 3-140-5 | Shipping Container Group, Standard Modules | Yes | Same | Yes | - | - | - |
| 3-140-6 | Work Platforms, Mobile | Yes | Same | Yes | Yes | - | - |
| 3-140-7 | Adapter Kit, Centaur/Shroud Transporter | No | Not required | - | - | - | - |
| 3-140-8 | Sling Assembly, Planetary Vehicle and Nose Fairing | No | Not required | - | - | - | - |
| 3-140-9 | Purge Unit, Freon/Ethylene Oxide | No | Not required | - | - | - | - |
| 3-140-10 | Planetary Vehicle Nose Fairing Mating and Assembly Fixture | No | Not required | - | - | - | - |
| 3-140-11 | Sling, Flight Capsule | No | Not required | - | - | - | - |
| 3-140-12 | Hoist Beam and Slings, Flight Spacecraft | No | Same | - | - | - | Yes |
| 3-140-13 | Tag Lines | Yes | Same | Yes | - | - | - |
| 3-140-14 | Platform, Launch Stand Access | No | Possibly | - | - | - | Yes |
| 3-140-15 | Universal Mounting Ring, Flight Spacecraft and Planetary Vehicle | No | Same | - | - | - | Yes |
| 3-140-16 | Environmental Cover, Flight Spacecraft | Yes or<br>No | Same<br>Same | Yes<br>- | Yes<br>- | -<br>- | -<br>Yes |
| 3-140-17 | Hoist Sling, Environmental Cover | Yes | Same | Yes | - | - | - |
| 3-140-18 | Platform, Auxiliary Access | Yes | Same | Yes | - | - | - |
| 3-140-19 | Transporter Adapter Cradle, 1969 Test Spacecraft* | - | 1969 only | - | - | - | Yes |

*Required for 1969 only.

spacecraft, which may differ from attach points on the 1971 flight spacecraft, but no equipment modifications are required.

### 2.2.4 Fixture, Weight, c.g., and MOI (TVS-3-140-4)

The same equipment functional requirements exists for the 1969 test spacecraft and the same design approach is used. If the 1971 fixture is used, the forward support ring, and the cradle assemblies will both require adapters to physically accept the 1969 test spacecraft mounting points. In addition, it is probable that different load cells will be employed because of the weight differences of the two spacecraft. The fixture balancing weights will be chosen and mounted at locations on the fixture consistent with the weight and physical arrangement and properties of the 1969 test spacecraft; therefore, they will probably not be the same weights or locations required for the 1971 flight spacecraft. The 1971 equipment will either be directly modified, or adapter kits will be provided.

### 2.2.5 Shipping Container Group Standard Module (TVS-3-140-5)

Since the 1969 test spacecraft will be used to test 1971 flight subsystem equipment, the same size standard modules require shipment and storage. The 1971 shipping container group will be used directly for the 1969 application with no modifications.

### 2.2.6 Work Platforms, Mobile (TVS-3-140-6)

The 1969 test spacecraft, mounted on the TVS-3-140-2 assembly, handling, and tilt fixture (modified) requires essentially the same access around the test spacecraft for assembly and test operations. The 1971 mobile work platforms are used directly, with few modifications if required. Modifications, as required, are provided in the form of modification kits, which allow restoration of the basic equipment to its original configuration.

2.2.7 Hoist Beam and Slings, Test Spacecraft (TVS-3-140-12)

The same requirements for hoisting and handling the 1969 test spacecraft exist, and in addition, the 1969 test spacecraft hoist beam and slings are used in hoisting and mating the test spacecraft to the launch vehicle at the launch pad. The design concept for the 1969 assembly is similar to the 1971 approach. However, the hoist beam assembly is designed to conform to the 1969 test spacecraft structural configuration. The hoist beam assembly is required to provide rigidity to the structure during these handling operations, and the slings attach directly to the hoist beam assembly.

2.2.8 Tag Lines (TVS-3-140-13)

The 1971 tag lines may be used directly for the 1969 test spacecraft in the same application with no modifications or alterations.

2.2.9 Platform, Launch Stand Access (TVS-3-140-14)

The 1969 launch stand access platform, if required, is specifically designed for the 1969 test spacecraft access requirements and the launch pad gantry tower configuration. The same design approach is employed, but the configurations are entirely different for the 1969 and 1971 spacecraft.

2.2.10 Universal Mounting Ring, Spacecraft (TVS-3-140-15)

A similar requirement exists for mounting the 1969 test spacecraft on various assembly and test fixtures and for protecting the mating flanges from damage. However, the 1969 test spacecraft requires its own mounting ring specially designed to the test spacecraft mounting face geometry. The design load requirements are less for the 1969 equipment.

2.2.11 Environmental Cover, Test Spacecraft (TVS-3-140-16)

The 1969 test spacecraft presents a high load profile when mounted in the VS-3-140-1 transporter (modified) than does the 1971 flight spacecraft. Therefore, a separate cover may be required. The design approach and requirements are identical to the 1971 equipment, but the size is increased to compensate for the higher profile. The cross-sectional shape of the cover may also require modification in order to conform to loading constraints of aircraft entry ramp and cargo compartment profiles. If the 1971 flight spacecraft environmental cover is designed to a modular panel fabrication concept, it may be possible to add additional panel modules in the fabrication process in order to provide the greater height required for the 1969 test spacecraft. If this is done, the 1969 cover is merely a modification of the 1971 cover, with complete restoration characteristics.

2.2.12 Hoist Sling, Environmental Cover (TVS-3-140-17)

The environmental cover for the 1969 test spacecraft requires a sling for handling. The design requirements and approach are essentially the same as for the 1971 equipment, and it is probable that the same sling will be used for both modules without modification.

2.2.13 Platform, Auxiliary Access (TVS-3-140-18)

The units selected or designed for the 1971 flight spacecraft system can be used directly for the 1969 test spacecraft system application.

2.2.14 Transporter, Adapter Cradle, 1969 Test Spacecraft (TVS-3-140-19)

Assuming that the same basic transporter transports both 1969 and 1971 spacecraft, an adapter cradle is required which will 1) mate to the upper and lower 1969 test spacecraft structural planes and 2) mount the test spacecraft in a horizontal position in the transporter. This adapter cradle carries the test spacecraft loads into the 1971 flight spacecraft mounting points and shock attenuation system in the transporter. This piece of MOSE is required for use only in the 1969 system.

## X. OSE SUBSYSTEM FUNCTIONAL DESCRIPTIONS

### 1. SUBSYSTEM ELECTRICAL OSE (Figure 10-1)

This section contains data on subsystem level EOSE, that is EOSE used to support subsystem level testing of the Voyager test spacecraft. The various unit test sets employed in production to assure acceptance performance levels of spacecraft subsystems are covered under their respective subsystems. Except for minor differences in panel details, this equipment is identical for both the 1969 and 1971 Voyager missions.

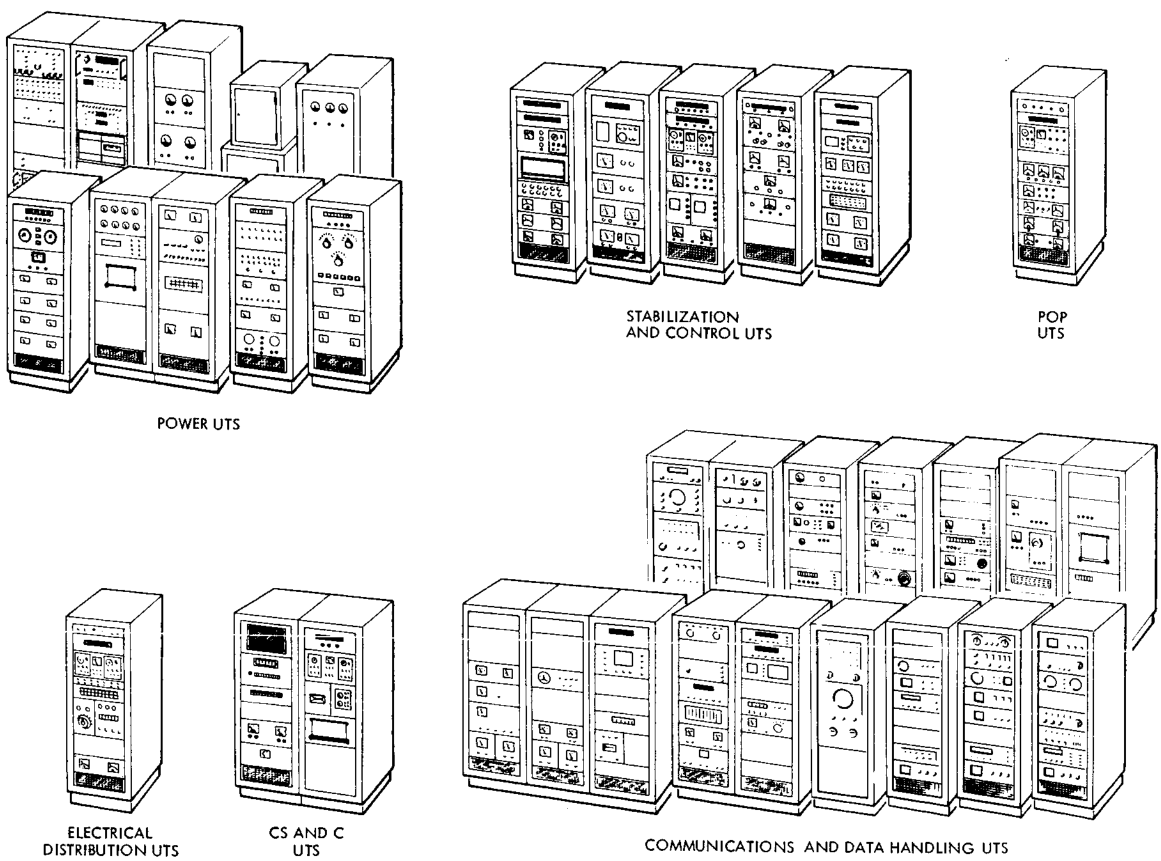



Figure 8. Electrical Operational Support Equipment

To obtain flexibility in scheduling, it is planned to perform integration assembly and testing on a panel level following qualification of units by unit test sets. This activity will be performed in a separate area apart from the spacecraft integration assembly and test area, and will be supported by a selection of unit test sets as appropriate rather than designing OSE specifically to duplicate the functions of the unit test set. Following is a discussion of the various unit test sets grouped under their respective spacecraft subsystems.

### 1.1 Communications and Data Handling Subsystem Unit Test Sets

These unit test sets are required to test and evaluate the Voyager spacecraft communications and data handling subsystem. They are used individually to test the associated flight units or collectively to test the integrated subsystem. The preliminary functional description of each required unit test set is contained in the following documents:

a) S-band communications unit test set—OSE/VS-4-311-1
b) VHF communications unit test set—OSE/VS-4-311-2
c) Command detector unit test set—OSE/VS-4-311-3
d) Data handling system unit test set—OSE/VS-4-311-4.

The antennas and coupling devices, both S-band and VHF, are mated to the integrated spacecraft and evaluated utilizing the system test set.

### 1.2 Stabilization and Control Subsystem Unit Test Sets

These unit test sets (UTS) are required to test the Voyager stabilization and control subsystem units, which consist of the rate gyro assembly, sun sensor and near earth detector, star sensors, control electronics assembly and actuators. Each of these units is provided with its own associated unit test set. The use of these unit test sets in one area can check out an integrated stabilization and control subsystem in the following modes:

a) Acquisition, cruise, re-orientation
b) Midcourse velocity correction
c) De-boost engine burn
d) Orbital operations.

The preliminary functional descriptions of the unit test sets are contained in the following documents:

a) Rate gyro assembly UTS - OSE/VS-4-411-1

b) Sun sensor and near earth detector UTS - OSE/VS-4-411-2

c) Star sensors UTS - OSE/VS-4-411-3

d) Stabilization and control electronics assembly UTS - OSE/VS-4-411-4

e) Actuator UTS - OSE/VS-4-411-5

### 1.3 Central Sequencing and Command Subsystem Unit Test Set

This unit test set is required to test the Voyager central sequencing and command subsystem (SC and C). Since the units comprising the CS and C subsystem are packaged into one integral unit, this unit test set provides the capability of testing the CS and C subsystem either as a unit prior to or after being mounted on its spacecraft panel.

The preliminary functional description of this test set is contained in OSE/VS-4-451-1.

### 1.4 Power Subsystem Unit Test Sets

The unit test sets required to test the Voyager power subsystem units consist of the main AC power inverter unit, the 410 cycle single phase inverter unit, the 820 cycle two phase inverter unit, the battery control unit, the power control electronics assembly (PCEA) and the battery unit. Because of the similarity of test requirements, capability of testing the three different inverter units is combined into the power inverter unit test set. Each of the other power subsystem units is tested by its own associated unit test set. The use of these unit test sets in one area can be used to check out an integrated power subsystem when mounted on the spacecraft panel.

The preliminary functional descriptions of each unit test are contained in the following documents:

a) Solar panel UTS - OSE/VS-4-461-1

b) Power inverter UTS - OSE/VS-4-461-2

c) Battery control UTS - OSE/VS-4-461-3

d) Power control electronics assembly UTS - OSE/VS-4-461-4

e) Battery UTS - OSE/VS-4-461-5

## 1.5 Electrical Distribution Subsystem Unit Test Set

This test set is required to test and evaluate the electrical distribution subsystem (EDS) either prior to or after its mounting onto the spacecraft panel. The preliminary functional description of the EDS unit test set is contained in OSE/VS-4-471-1.

## 1.6 Planet Oriented Package Unit Test Set

This section contains the preliminary functional specification for this unit test set required to test the Voyager spacecraft planet oriented package (POP). The test sets required to test the other scientific experiments aboard the spacecraft are provided as GFE along with the scientific experiment. The POP unit test set provides the capability of testing the Mars sensor, gimbal drive and pickoff, and the gimbal electronics portions of POP in the alignment and servo modes.

The preliminary functional description of this test is contained in OSE/VS-4-581-1.

## 1.7 Propulsion Subsystem Unit Test Set

Analysis of the test requirements for the propulsion subsystem disclosed that no electrical unit test sets are required. Functional operation and verification of the propulsion subsystem electrical components (valves, feedback pots, etc.) will be made during integrated systems test by the system test set.

# 2. SUBSYSTEM MECHANICAL OSE

## 2.1 General

This section defines the equipment required for the assembly, alignment handling, protection, transport, shipping, and storage of the 1969 test spacecraft subsystems. Because of equipment similarities and the desire to use common mechanical operational support equipment for both the 1969 test spacecraft subsystems and the 1971 flight spacecraft subsystems, the referenced OSE is related to and must be interpreted with the following documents and their related equipment documents included in Voyager Phase IA Study Report, Volume 6, Section IV.

a) OSE/VS-4-210 1971 Voyager Mechanical Operational Support Equipment, Science Payload Subsystem

b) OSE/VS-4-310 1971 Voyager Mechanical Operational Support Equipment, Communications and Data Handling Subsystem

c) OSE/VS-4-410 1971 Voyager Mechanical Operational Support Equipment, Stabilization and Control Subsystem

d) OSE/VS-4-460 1971 Voyager Mechanical Operational Support Equipment, Power Subsystem

e) OSE/VS-4-510 1971 Voyager Mechanical Operational Support Equipment, Thermal Control Subsystem

f) OSE/VS-4-520 1971 Voyager Mechanical Operational Support Equipment, Structural Subsystem

g) OSE/VS-4-530 1971 Voyager Mechanical Operational Support Equipment, Pyrotechnic Subsystem

h) OSE/VS-4-580 1971 Voyager Mechanical Operational Support Equipment, Planet Oriented Package Subsystem

i) OSE/VS-4-610 1971 Voyager Mechanical Operational Support Equipment, Propulsion Subsystem

To distinguish differences between the 1969 and 1971 Voyager mission OSE where they exist, the equipment described in this section is identified as follows:

a) OSE/TVS-4-310 Series (Communication and Data Handling Subsystem)

b) OSE/TVS-4-410 Series (Stabilization and Control Subsystem)

c) OSE/TVS-4-460 Series (Power Subsystem)

d) OSE/TVS-4-510 Series (Thermal Control Subsystem)

e) OSE/TVS-4-520 Series (Structures Subsystems)

f) OSE/TVS-4-530 Series (Pyrotechnic Subsystem)

g) OSE/TVS-4-610 Series (Propulsion Subsystem)

All applicable documents, general requirements, and specific equipment requirements are the same for both the 1969 and 1971 mechanical operational support equipment (MOSE) except as noted. The equipment requirements for the 1969 and 1971 systems are compared in the 1969/1971 MOSE Comparison Matrix shown in Table II.

Following is a series of notes indicating differences between OSE for the 1969 and 1971 Voyager missions. During Phase IB full specifications will be generated for this equipment.

## 2.2 Equipment Description

### 2.2.1 Dolly, 6 Foot Elliptical Parabolic Antenna (TVS-4-310-1)

The 1969 test spacecraft elliptical parabolic antenna is the same size as the 1971 antenna. The 1971 antenna dolly adequately supports the 1969 antenna. The design requirements are the same. However, a modification kit for the dolly is required to mount the 1969 antenna hoist beam, which is a 1969 item.

### 2.2.2 Hoist Beam, 6 Foot Elliptical Parabolic Antenna (TVS-4-310-2)

The 1969 elliptical parabolic antenna is supported on the test spacecraft and actuated during deployment in a fashion which precludes usage of similar mounting and deployment arm structure. Therefore, the 1969 antenna hoist beam is a new design especially configured to the antenna configuration. However, the design requirements remain essentially the same.

### 2.2.3 Shipping Container, 6 Foot Elliptical Parabolic Antenna (TVS-4-310-4)

The 1971 antenna shipping container is suitable for use with the 1969 antenna. The design requirements are essentially identical. However, the foam encapsulation material is die cut to the different antenna deployment arm configuration, and is provided as a modification kit.

### 2.2.4 Shipping Container, Low Gain Antenna (TVS-4-310-5)

The 1969 Low Gain antenna is the same configuration and size as its 1971 counterpart. The shipping container for this antenna has the same design requirements. The foam encapsulation material is die-cut to the 1969 antenna mounting arm configuration, and the container may be used for both systems.

2.2.5 Alignment Fixture, Stabilization and Control Nozzles (TVS-4-410-1)

The same alignment fixture may be used directly for both the 1969 test spacecraft nozzles and the 1971 flight spacecraft nozzles.

2.2.6 Protective Covers, Stabilization and Control Nozzles (TVS-4-410-2)

The mounting arrangement and structure for the 1969 test spacecraft stabilization and control nozzles, as well as the nozzle blocks themselves, will probably differ enough from the 1971 flight spacecraft configuration to preclude usage of the 1971 protective covers. Since the equipment is very inexpensive, special 1969 covers will be designed, using the same design requirements and basic design approach.

2.2.7 Assembly and Handling Frame, Solar Panel Segment (TVS-4-460-1) (TVS-4-460-1)

2.2.8 Protective Cover, Solar Panel Segment (TVS-4-460-2)

2.2.9 Shipping Container, Solar Panel Segment (TVS-4-460-3)

2.2.10 Handling Dolly, Solar Panel Segment (TVS-4-460-4)

2.2.11 Sling, Assembly, Solar Panel Segment (TVS-4-460-5)

This entire equipment group (2.7 through 2.11) is designed specifically for the 1969 test spacecraft power subsystem because the 1969 solar panel segments are of different size and shape. The design requirements for the 1969 equipment group are essentially the same as those for the 1971 equipment group. The 1969 test spacecraft panels are wedge shaped, approximately 100 inches long, and 54 inches wide at the base, tapering to 38 inches wide at the outboard edge. There may also be a bend or kink in the panel plane, so that the entire panel segment consists of two intersecting plane sections. Although the design approach to the handling equipment is similar to the 1971 equipment, the equipment is configured to the unique panel shape, and structural support is designed to meet panel peculiarities.

### 2.2.12 Shipping Container, Battery (TVS-4-460-6)

The 1969 flight batteries will probably be identical to the 1971 batteries. The 1971 shipping container may be used in direct support of the 1969 batteries.

### 2.2.13 Shipping Container, Power Amplifier (TVS-4-460-7)

The 1969 test spacecraft is equipped with the same power amplifiers to be used in 1971. Therefore, the same shipping container is used for both series.

### 2.2.14 Assembly and Handling Fixture, Spacecraft Louvers (TVS-4-510-1)

The 1969 test spacecraft is equipped with the same spacecraft thermal control louvers which the 1971 flight spacecraft will use. Therefore, the same fixture may be used for both series.

### 2.2.15 Shipping Container, Spacecraft Louvers (TVS-4-510-2)

Since the same spacecraft louvers are used on both the 1969 and 1971 spacecraft, the same shipping container is employed for both series.

### 2.2.16 Handling and Shipping Container, Insulation (TVS-4-510-3)

The 1969 test spacecraft is equipped with the same type of insulation for the spacecraft panels, etc. It is packaged, handled, and shipped in the same fashion as the 1971 insulation. Although the sizes of panel sheets vary, the packaging concept developed for 1971, and the equipment configuration proposed will adequately support both missions. The 1971 handling and shipping container may be used in 1969 with no changes or modifications.

### 2.2.17 Dolly, Structural Sections (TVS-4-520-1)

Since this dolly is relatively simple and inexpensive, modifying it to accept 1969 test spacecraft structural assemblies by providing modification kits to the 1971 design would probably cost as much as providing a separate dolly for the 1969 test spacecraft structural elements. Therefore a new dolly, based on the same design and functional requirements and using the same design approach, is recommended.

2.2.18 Shipping Container, Miscellaneous Spacecraft Structure (TVS-4-520-2)

Since the 1969 test spacecraft miscellaneous structure differs from the 1971 flight spacecraft structure, and since the shipping container consists merely of foam chocks and an outer wooden container, there appears to be no reason to try to adapt the 1971 container to 1969 use. Therefore a 1969 container is separately provided based on the same design approach as for 1971.

2.2.19 Sling, Propulsion/Pneumatic Structural Section (TVS-4-520-3)

The different structural configurations of the 1969 test spacecraft suggest the use of a new sling assembly, using a four leg assembly to attach to the four corners of the spacecraft bus structural section. The same design requirements and approach are used, but the item is designed for use with the 1969 test spacecraft only.

2.2.20 Interface Match Tool, Spacecraft/Centaur Adapter (TVS-4-520-5)

The 1969 test spacecraft has a different structural interface configuration with the Centaur adapter than the 1971 flight spacecraft. Therefore, a new interface match tool, with the same functional and design requirements, and design approach is required for 1969.

2.2.21 Handling Case, Arming Kit (TVS-4-530-2)

An arming kit handling case is required for the 1969 test spacecraft for final arming and installation of category A squibs and detonators. The case has the same functional and design requirements as the 1971 spacecraft arming kit. Since the number of squibs and detonators, and possibly their type, differs in 1969, the die-cut foam pad insert to the handling case will be appropriately fabricated for the 1969 series and inserted in the case. The 1971 pad insert replaces it for the later series. All other subassemblies will be identical.

2.2.22 Alignment Fixture, Midcourse Engine (TVS-4-610-4)

The same midcourse engine is used in both the 1969 test spacecraft and the 1971 flight spacecraft. The same alignment fixture is used for both series.

2.2.23 Shipping Container, Midcourse Engine (TVS-4-610-6)

The same midcourse engine is used on both the 1969 and 1971 spacecraft. Therefore the shipping equipment is identical, and very possibly the same unit.

2.2.24 Pneumatic Test Set (TVS-4-610-7)

The same pneumatic test set is used for both the 1969 and 1971 spacecraft.

2.2.25 Pneumatic Fill Cart (TVS-4-610-8)

Although the quantities of helium and nitrogen required for the 1969 spacecraft are less than for the 1971 spacecraft, the same pneumatic fill cart is used for both spacecraft.

2.2.26 Propellant Transfer and Handling Cart (TVS-4-610-9)

Although the propellant quantity required for the 1969 spacecraft is less than for the 1971 spacecraft, the same propellant transfer and handling cart is used for both spacecraft.

2.2.27 Alignment Fixture, Midcourse Engine/Steering Vanes (TVS-4-460-10)

Since the same midcourse engine is used for both the 1969 test spacecraft and the 1971 flight spacecraft, the same steering vanes alignment fixture may be used for both series.

2.2.28 Universal Handling Fixture, Hydrazine/Helium Tank (TVS-4-610-11)

The 1969 test spacecraft is equipped with one hydrazine/helium tank of identical dimensions and materials to the 1971 flight spacecraft tanks. Therefore the same universal handling fixture is used with no modifications.

2.2.29 Sling, Hydrazine/Helium Tank (TVS-4-610-12)

Since both 1969 and 1971 spacecrafts are equipped with the same size tanks, the same sling assembly is also used.

# XI. OSE IMPLEMENTATION PLAN

## 1. INTRODUCTION

Due to the similarities in OSE in the 1969 and 1971 Voyager missions, the OSE implementation plan contained in Volume 6, Section V for the 1971 Voyager mission covers both the 1969 and 1971 OSE development. This implementation plan identifies various activities required in Phase IB and Phase II to accomplish the development of the operational support equipment required in support of both Voyager missions. To provide an overall impression of the 1969 Voyager OSE effort, the major milestone schedule and tabulations of equipment quantities are provided below.

## 2. OSE SCHEDULE

The major milestone schedule which is printed in Section V of Volume 6 indicates the milestone requirements scheduled for both the 1969 and 1971 Voyager missions, and indicates that deliveries of AHSE meet requirements for assembly of the 1969 and 1971 engineering models; the unit test sets meet requirements for spacecraft compatibility and type approval tests; the system test sets, automatic data handling system and launch complex equipment meet requirements for assembly of both engineering models and proof test models, and the mission dependent equipment meets requirements for spacecraft compatibility tests with the deep space information network.

## 3. OSE QUANTITIES AND LOCATIONS

Data on OSE quantities required to support the 1969 Voyager test mission will be found combined with the 1971 data in Section V of Volume 5 of this Study Report.

AUG 12 1965

SIGNIFICANT ERRATA. TRW Systems, Phase 1A
Study Report, Voyager Spacecraft
August 11, 1965

Volume 1. Summary

Substitute new p. 79 attached.

Volume 2. 1971 Voyager Spacecraft

p. 18. Item h) "necessary landed operations" should read "necessary lander operations."

p. 143. Section 3.4.1.a. second line should read "threshold of 0.25 gamma"

p. 282. Lines 3 and 4. Delete "or incorrect spacecraft address"

p. 284. Figure 5. Change "128 Word DRO Core Memory" to "256 Word DRO Core Memory"

p. 327. Denominator of second term on right hand side of equation should read

$$\left(\frac{1}{\epsilon_1} + \frac{1}{\epsilon_2} - 1\right)\left(N - 1\right)$$

p. 351. Figure 1, Section F-F. "separation nut" should read "bolt catcher"

Volume 3. Voyager Program Plan

Substitute new p. 12 attached.

p. 13. Figure 2-3. PTM Assemblies in item 7 move 1.5 months to right

p. 16. Figure 2-6. First milestone date should be September 1, 1969, instead of mid-January 1970, and all subsequent dates should be correspondingly adjusted 4.5 months earlier.

p. 20. Table 2-2. Third item in 1969 column should read "coincident with completion of proof test model assemblies. Fifth item in this column change "2 weeks" to "3.5 months." Fourth item in 1971 column, change "4 months" to "5 months."

p. 67. Figure 5-2. Under Intersystem Interface Specification add a block entitled "Spacecraft to OSE Interface Specification"

p. 120. Last line of paragraph c should read "shown in Table 5-2."

p. 126. Figure 5-13. Year should be 1966 instead of 1965.

p. 153. Figure 5-18. Ignore all numbers associated with lines in figure.

p. 167. Figure 5-21. In line 20 change "design revisions" to "design reviews"

p. 254. Second paragraph, third line, "The capability of the transmitter to select" should read "The capability of the transmitter selector to select."

p. 258. Section heading n should read <u>Experiment Data Handling</u>

p. 604. Section 3.2.1 beginning of second paragraph should read "The hydrazine fuel . . . "

## Volume 4. <u>Alternate Designs: Systems Considerations</u>

p. 103. Figure 3-19. Caption should read "Radial Center of Mass. . . "

p. 151. Last paragraph, second line, "For the baseline, the reliability. . ." should read "The reliability . . . "

p. 158. 8th line, replace "0.06 pound/watt" by "0.6 pound/watt"

p. 215. Figure 3-50. Dot in ellipse at right should be 0.

p. 230. Section 5.3.2, second paragraph, 7th line, should read "Figure 3-52."

p. 239. Second line, "with a variable V" should read "with a variable $\Delta V$"

p. 247. First line, "3250 km/sec" should read "3.250 km/sec"

p. 261. Figure 3-64. Interchange coordinates, clock angle and cone angle

p. 293. Figure 3-81. An arrow should connect "Low-gain spacecraft antenna" and the dashed line at $73 \times 10^6$ km

## Volume 4. <u>Alternate Designs: Systems Considerations Appendix</u>

p. 16. Figure A-2. The shaded portion under the lower curve should extend to the right only as far as 325 lb.

p. 9. Table A-1, part (1). In last column heading change "$W_3$" to "$W_1$". In part (4) last column heading change "$W_3$" to "$W_4$"

p. 22. Second line below tabulation, replace "575 × 35" by "570 × 35"

p. 29. Tabulation at bottom of page, change "18" to "30" and "400" to "240"

p. 207. Numerator of equation for $\lambda$ best at bottom of page should read "0.0201," and numerator of equation for $\lambda$ worst should read "9.21"

p. 209. Table 5B, fifth line. Delete "$\times 10^{-}$." Also p. 213, Table 7A, seventh line, and p. 232, Table 3B, fifth line.

p. 217. Top portion of Table 9B should be labeled "primary mode" instead of "other modes"

p. 326. In equations following words "clearly" and "thus" insert "$>$" before second summation.

Volume 5. Alternate Designs: Subsystem Considerations

p. 3-15 Fifth line, "... is extended, spacecraft" should read "... is extended, two spacecraft"

p. 3-38 Last line, change " $= \frac{32}{4500} = M$" to "$\left(\frac{32}{4500}\right)\left(M\right)$"

p. 3-51 Two equations at bottom of page should read

$$D = 4\pi A/\lambda^2$$

$$A = \frac{D\lambda^2}{4\pi} = \frac{1000\,\lambda^2}{4\pi}$$

p. 3-67 Third line, last parenthesis "$\left(\frac{\pi}{2} + \phi\right) -$"

p. 3-82 6th line should read "50 degrees" instead of "50-140 degrees," and seventh line should read "140 degrees" instead of "50-140 degrees"

p. 3-111 Last line, change "50 Mc" to "1 Mc"

p. 3-137 Item g) for "... followed by 5 frames of real time" substitute "... followed by 11 frames of low rate science data and 5 frames of real time" pp. 3-150 and 3-151 are interchanged.

p. 3-156 Last line, should read "gates, a 7 bit"

p. 5-21 Second paragraph, third line, for "others since they are" substitute "others which are"

p. 5-33 Bjork equations should identify 0.18 as an exponent, and the exponent for $(\rho_p/\rho_t)$ in the Hermann and Jones equation should be 2/3 in both cases.

p. 5-33 Figure 5-12 should be replaced with Figure C-7 of Appendix C.

p. 5-40 Three lines above Table 5-10 substitute "permanent set" for "experiment"

Volume 5. <u>Alternate Designs: Subsystem Considerations. Appendix I</u>

p. B-11 Bottom of page, for "$r^{2/3}$" substitute "$(V/C)^{2/3}\ r$"

p. C-4 The title of Figure C-2 should read "Figure C-2. Meteoroid Influx Rate Circular Orbit Mars", and the title of Figure C-3 should read "Figure C-3. Meteoroid Influx Rate Cruise"

p. C-5 At bottom of page, add the following: "*Within 50,000 km of Mars"

p. C-6 Line 13 should read: "... of low density ($\rho_p < 2.4$ gm/cm$^3$..."

p. C-6 Figure C-4. The ordinate "2" should read "100"

pp. C-17 C-21 The figures C-6 and C-7 on pages C-17 and C-21 should be reversed.

p. C-28 The title of Figure C-8 should read "Meteoroid Shield Test Specimen"

p. C-29 The title of Figure C-9 should read "Cutaway of Meteoroid Shield Test Specimen

p. C-34 In Section 1.8 the first sentence should be replaced by the following two sentences: "Preceding sections of this appendix contain derivations of the probability of penetrations of the spacecraft outer skin by meteoroids. It is clear that to design an outer skin of sufficient thickness to reduce the probability of no penetrations to a low level, such as 0.05 to 0.01, would be prohibitive in terms of the weight required."

p. C-35 In the first equation, the expression "(t in $m^2$)" in two places should read "(t in cm)" and "A" in two places should read "(A in $m^2$)"

p. C-38 In Table C-2, all values in inches should be in centimeters. A zero should be inserted immediately following the decimal point, for example: (0.020-inch) = 0.05080, (0.020-inch) = 0.06096, (0.020-inch) = 0.04064, etc.

p. C-40 In Section 1.8.7 Computation of $R_i'$, the sixth line should read "... than $10^{\circ}$ are neglected"

p. C-45 In listing under "Values of t Used for Extreme Environment Analysis," under Inch, the first number should read 0.020 instead of 0.202

p. C-52 In 1.10 NOMENCLATURE, "$K_2$" should be defined as "$K^{-2/3}$ (4 ±2)" and "B" should be

$$\frac{1000\ \rho_t V^2}{9.806\ H_t}$$

pp. C-150 and C-151 should be reversed.

p. C-208 Along the ordinate in the graph, "Stress × $10^{-3}$" should read "Stress × $10^{-2}$"

Volume 5. Alternate Designs: Subsystem Considerations. Appendix II

p. F-23 Lines 7 and 10 change all subscript $\tau$ to T

p. F-24 Line 14, change "$ME_1$" to "$mE_1$"

p. F-29 Figure F-9 title should be "Reflection Phase Angle $\phi$ (deg)" and Figure F-10 title should be "Reflection Magnitude R"

p. F-30 Last line, change "0.27" to "0.175"

p. F-31 Lines 14 and 15, change "14,700 ft/sec to 460 ft/sec" to 14,700 ft/sec minus 460 ft/sec" and "14,700 ft/sec to 10,000 ft/sec" to "14,700 ft/sec minus 10,000 ft/sec"

p. F-32 Last line in item 4), change "27 per cent" to "17.5 per cent"

p. F-35 Table F-4, under Assumed Parameter for item 2 insert "±2 × $10^{-5}$", for item 3 insert "±3 × $10^{-5}$", and for item 4 insert "±2 × $10^{-5}$"

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|---|---|
| p. F-53 | Item d. Noise Figure, change "4 db" to "3.5 db"; Gain, change "20 db" to "10 db", last line change "10 db" to "4 db" |
| p. F-58 | Figure F-21. Change 102 kc to 112 kc. |
| p. F-59 | Line 22, change to "$M_1$ = 21.5 deg or 0.375 radians (rms, peak)" |
| p. F-60 | Line 2, change to |

$$"M_2 = \sqrt{(1.1)^2 - (0.375)^2}"$$

| | |
|---|---|
| p. F-60 | Line 3, change to "$M_2$ = 1.03 radians (rms) or 1.46 radians (peak)" |
| p. G-6 | Paragraph 1.4, second line, change "from $E_M = 10^1 E_o$ to $10^4 E_o$ . . ." to read "from $E_M = 10^{-1} E_o$ to $10^4 E_o$ . . ." |

Volume 6. Operational Support Equipment

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|---|---|
| p. 25 | Figure 6. Caption should be "Typical Grounding Scheme" |
| p. 39 | Section 1.3.3, change opening of first sentence to read "Launch pad equipment consists of the ground power and RF consoles and the test flight program power and control equipment . . ." |
| p. G-31 | Figure 1. Lines enclosing Data Format Generator should be solid. |
| p. G-102 | Last line substitute "4500" for "45" |
| p. G-113 | In Section 4.4.2, change "25 per cent" to "250 per cent" |
| p. G-134 | Section 4.5, substitute "6.5 feet" for "six feet" |
| p. G-311 | Fifth line, change "30 per cent" to "20 per cent" |
| p. G-398 | Section 4.2 should begin with "The hoist beam is . . ." |
| p. G-419 | Second line "4 optical alignment targets" instead of 8. Same correction top of p. G-421. |
| p. G-423 | Section 4.9.2, substitute "20 per cent" for "50 per cent" |

Volume 7. <u>1969 Flight Test Spacecraft and OSE</u>

p. 90 First line should read "Launch pad equipment consists of the ground power and RF consoles and . . . "

p. 107 Last line, change Volume 5 to Volume 6.